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SIMULATOR STUDY OF ABILITY OF PILOTS TO ESTABLISH NEAR-CIRCULAR LUNAR ORBITS USING SIMPLIFIED GUIDANCE TECHNIQUES

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SUMMARY

A study has been made with a six-degree-of-freedom fixed-base simulator of the ability of pilots to establish 486 000-foot (80-nautical-mile) circular orbits about the moon using a simplified guidance technique. The pilot had control of thrust along the longitudinal axis and of vehicle attitude through an acceleration command system. No automatic damping or control was assumed. The general guidance procedure consisted of maintaining a constant thrust angle with respect to the lunar horizon until attaining the proper altitude with zero radial velocity. A constant-altitude deceleration maneuver was then performed to attain the proper circumferential velocity. Initially, a "nominal trajectory" was flown for which the exact operating procedure was specified; this trajectory was followed by several off-nominal trajectories for which no operating procedure was specified.

The results of the investigation showed that if velocity and altitude information were available, the pilots could consistently establish orbits lying within an altitude range from 361 000 to 611 000 feet by using rather crude thrust angle measurements. The characteristic velocity required to perform the maneuver was within 5 percent of that required for a perfectly flown nominal.

The pilots could consistently establish circular orbits from the nominal-approach trajectory in the absence of velocity and altitude information. This condition was accomplished by maintaining a constant thrust angle with respect to the lunar horizon for a given time followed by a second constant thrust angle for a second specified time. This procedure did not result in the establishment of near-circular orbits when used under the influence of possible earth-based tracking errors but did result in establishing nonimpacting orbits.

INTRODUCTION

One of the present concepts (ref. 1) for accomplishing the lunar landing mission involves establishing a circular orbit about the moon at an altitude of approximately 486 000 feet. Injection into the circular orbit is to be performed with an automatic guidance and control system. However, manual procedures can be developed to be used as a backup mode, or, if sufficiently precise and simple, might be considered as a primary control mode.

A simulation study of a manual control procedure for establishing lunar orbits (ref. 2) indicated that near-circular orbits could be established. The piloting procedure in reference 2 was very simple; however, the display used for guidance would be rather complex for a spacecraft. It is, therefore, desirable to formulate simple guidance and control procedures that require as little display information as possible. The present fixed-based simulation study, which used a "hybrid" digital-analog computer, was to investigate the ability of pilots to establish 486 000-foot circular orbits about the moon by using a simplified guidance scheme.

It is assumed that the vehicle has been injected toward the moon on a trajectory that results in an altitude at pericynthion slightly in excess of 486 000 feet. The pilot's task is to apply retrothrust, the lunar horizon being used as a visual reference, to attain pericynthion at an altitude of 486 000 feet and to reduce velocity to that of a circular orbit at that altitude.

SYMBOLS

Any consistent set of units may be used. In this report it is assumed that

1 international foot = 0.3048 meter
1 international nautical mile = 1.852 kilometers

F	thrust, pounds
g_e	acceleration at surface of earth due to gravitational attraction, 32.2 feet/second^2
g_m	acceleration at lunar surface due to gravitational attraction, 5.32 feet/second^2
h	altitude above lunar surface, feet
I_{sp}	specific impulse, 313 seconds
I_X, I_Y, I_Z	moments of inertia about the vehicle X-, Y-, and Z-axes, respectively, $\left(I_X = I_Y\right)$, slug-feet^2
K	angle between thrust vector and line of sight to lunar horizon, positive above horizon, radians or degrees

M_X, M_Y, M_Z	control moments exerted about vehicle X-, Y-, and Z-axes, respectively, foot-pound
m	vehicle mass, slugs
$\dot{m}$	time rate of fuel consumption, slugs/second
p, q, r	rates of rotation about vehicle X-, Y-, and Z-axes, respectively, radians/second
R	radial distance from center of moon, feet
R_m	radius of moon, 5.702×10^6 feet
$\dot{R}$	vehicle velocity component in radial direction, feet/second
$R\dot{\theta}$	vehicle velocity component in circumferential direction, feet/second
t	time, second
ΔV	characteristic velocity, $g_e I_{sp} \log_e \frac{m_o}{m}$, feet/second
W	earth weight of vehicle, mg_e, pound
X, Y, Z	vehicle body-axis system with origin located at vehicle instantaneous center of gravity and with Z-axis alined with vehicle axis of symmetry
x_i, y_i, z_i	inertial reference axes with origin located at lunar center (fig. 1)
x, y, z	distances along the x_i-, y_i-, and z_i-axes, respectively, feet
x', y', z'	reference axes parallel to inertial axes with origin located at vehicle center of gravity
Δz	vehicle lateral displacement with respect to initial orbit plane, feet
δ_F	rocket throttle control displacement
$\delta_X, \delta_Y, \delta_Z$	control displacements which produce control moments about X-, Y-, and Z-axes, respectively
ψ, θ, ϕ	Euler angles of rotation, radians or degrees
$\bar{\psi}$	angular orientation of vehicle referred to as yaw angle and defined as angle between vehicle thrust axis and trajectory plane, radians or degrees

$\bar{\phi}$ bank angle, radians or degrees

$\bar{\theta}$ angular travel over lunar surface, radians or degrees

Subscripts:

a pertains to conditions at apocynthion

o initial conditions

p pertains to conditions at pericynthion

A dot over symbols denotes differentiation with respect to time.

EQUATIONS OF MOTION

The basic equations of motion used in this study permitted six rigid-body degrees of freedom and are given with pertinent auxiliary equations in the appendix. The three force equations were written with respect to an inertial axis system; the three moment equations were written with respect to the vehicle body axes. The pilot closed the control loop and had direct input into the force and moment equations. The equations of motion were solved on a hybrid digital-analog computer operating in real time.

The inertial axis system for the force equations was a fixed-axis system with its origin at the center of the moon (fig. 1) which was assumed to be a nonrotating homogeneous sphere. Vehicle mass and moments of inertia were varied as thrust was applied to account for mass reduction during thrusting. Mass changes due to the use of moment controls were negligibly small compared with the change associated with thrust application and were neglected.

VEHICLE DESCRIPTION

The vehicle assumed in this study was a body of revolution with the crew compartment at the nose. The vehicle had a single fixed engine which thrusted along the axis of symmetry. Thrust was either applied at the maximum level or was off. The maximum thrust level chosen for the study resulted in accelerating the vehicle at thrust initiation at $0.28g_e\left(\frac{F}{W_o} = 0.28\right)$. Upon orbit attainment, the acceleration due to thrust was approximately $0.38g_e$ as a result of vehicle mass reduction.

An acceleration command system which provided moment control about the vehicle body axes was assumed to be generated by reaction jets operating in pairs to produce pure torques. The variation of the vehicle moments of inertia with vehicle mass is presented in figure 2.

Cockpit and Controls

The general layout of the cockpit is presented in figure 3 and shows the relative position of the pilot's seat with respect to the controls and instrument display. Vehicle thrust was commanded by the use of the displacement controller located at the pilot's left. Moment control about the three body axes was commanded through the three-axis hand controller located to the right of the pilot. The controller had maximum angular displacements about each of its axes of approximately 40°. The control torques commanded by the pilot for attitude control were proportional to control deflection except for a small deadband around zero deflection. (See fig. 4.)

Instrument Display

The display and controller motions were consistent with the pilot looking along the axis of symmetry in the direction of thrust. The instrument display shown in figure 3 is a general-purpose display and includes quantities not used in the present study. A sketch of the instrumentation used in this investigation is presented in figure 5. The altimeter and the radial and circumferential velocity meters were digital readout instruments. The resolution of these meters was such that altitude could be read to within 500 feet, radial velocity to 1 foot per second, and circumferential velocity to 10 feet per second. Vehicle out-of-plane velocity and displacement were given on galvanometer-type meters with resolutions of 25 feet per second and 2500 feet, respectively.

Vehicle attitude and attitude rates were given on a three-axis ball indicator. The angle K between the vehicle thrust vector and the line of sight to the lunar horizon was presented on a separate meter which could be read to 1° and estimated to within 0.5°.

Located above the instrument panel was an oscilloscope presentation of three "stars" and the "lunar horizon." This view corresponds to that observed looking through a telescope or a window directed along the axis of symmetry in the direction of thrust. The assumed window or telescope has a field of view of $\pm 30^\circ$ and a reticle which provided a resolution in pitch and yaw of about 6°. However, pitch and yaw could be estimated to within about 1°. The initial flights of the investigation used the K-meter and ball indicator combination for attitude reference. However, in most of the flights the oscilloscope presentation was used for attitude information in which case the K-meter was covered and the ball indicator was caged.

LUNAR APPROACH TRAJECTORIES

Nominal Trajectory

The nominal lunar approach trajectory used in the present investigation had an altitude of 499 000 feet and a velocity of 8300 feet per second at pericynthion and is typical of trajectories currently being considered for the lunar mission. The vehicle thrust level gave an initial deceleration of $0.28g_e$. Thrust was applied when the altitude above the lunar surface reached 580 000 feet and was terminated at an altitude of 486 000 feet.

Off-Nominal Trajectories

The off-nominal trajectories differed from the nominal such as might be expected because of small errors remaining after midcourse corrections. The assumed variations about the nominal conditions at problem initiation were ±50 feet per second for the radial and circumferential velocities and ±10 000 feet for altitude. Additional off-nominal conditions of ±100 000 feet about the nominal initial altitude were considered. These values correspond to the errors that might exist if the altitude was obtained from earth-based tracking systems. Errors in velocity due to the use of earth-based tracking lie within the ±50 feet per second previously mentioned.

The following table lists the initial conditions for the various types of trajectories flown during the investigation:

Radial velocity, $\dot{R}$, ft/sec	Circumferential velocity, $R\dot{\theta}$, ft/sec	Altitude, h, ft	Remarks
-1 138	8 125	601 600	Nominal trajectory
			Off-nominal trajectories
-1 088	8 125	601 600	Low in $-\dot{R}$
-1 188	8 125	601 600	High in $-\dot{R}$
-1 138	8 175	601 600	High in $R\dot{\theta}$
-1 138	8 075	601 600	Low in $R\dot{\theta}$
-1 138	8 125	611 600	High in h
-1 138	8 125	591 600	Low in h
-1 188	8 075	591 600	High in $-\dot{R}$, low in $R\dot{\theta}$ and h
-1 138	8 125	701 600	High in h
-1 138	8 125	501 600	Low in h

GUIDANCE SCHEME

Analysis

Two situations were considered in developing the simplified guidance scheme. In one case it was assumed that the information available to the pilot included altitude and the vehicle velocity components in the radial, circumferential, and out-of-plane directions. Such information might be obtained with an onboard radar system. In the second case, it was assumed that any information available to the pilot was obtained from an earth-based tracking system.

Preliminary investigation indicated that the angle between the thrust vector and the line of sight to the lunar horizon varied considerably along the gravity-turn trajectory used to inject into a 486 000-foot circular orbit. However, it was found that there was a constant value of the thrust angle with respect to the lunar horizon which, if maintained, would result in placing the vehicle at the proper altitude with zero radial velocity. If it is assumed that the radar system was operative, the pilot could then perform a constant-altitude deceleration maneuver until the proper circumferential velocity was attained. Thus it was felt that the lunar horizon could provide a convenient reference for thrust-vector orientation in the formulation of a simplified guidance technique. A sketch of the trajectory relative to the lunar surface is shown in figure 6. In the event that the radar system was inoperative, the constant thrust angle would be maintained for a specified amount of time and the constant-altitude deceleration maneuver would be approximated by maintaining a second constant thrust angle for a given time.

An analytical investigation was performed by using an electronic digital computer to determine the proper values of the constant thrust angles and thrusting times used in the guidance scheme. It was found that an initial thrust angle of 28° placed the vehicle at an altitude of 486 000 feet with zero radial velocity when the thrust angle was maintained for 217 seconds. Maintaining a second constant thrust angle of 19° for 69 seconds reduced the circumferential velocity to the proper level for a circular orbit at that altitude. Figure 7 presents a hodograph of the nominal trajectory. The characteristic velocity required to perform this maneuver was about 2980 feet per second.

Piloting Procedures

As indicated previously, the pilot's task was to establish a circular orbit about the moon at an altitude of 486 000 feet. To accomplish this orbit, the pilot must modify the lunar approach trajectory to reduce the vehicle velocity components to $\dot{R} = 0$ and $R\dot{\theta} = 5285$ feet per second at that altitude. If the radar system is working, the pilot observes velocity and altitude to determine whether he is on the nominal trajectory. If nominal conditions prevail, the pilot directs thrust 28° above the lunar horizon until the vehicle descends to about 486 000 feet at which time he adjusts pitch so as to obtain zero rate of descent at that altitude. The pilot then manipulates the vehicle pitch attitude in order to maintain $\dot{R} = 0$ until $R\dot{\theta}$ is reduced to 5285 feet per second. In the event that off-nominal conditions prevail, the pilot adjusts the time of thrust initiation and thrust angle in an attempt to reduce the effect of the off-nominal condition. As the vehicle approaches the proper altitude, the pilot maneuvers to bring $\dot{R}$ to zero and performs a constant-altitude deceleration maneuver to obtain the proper value of $R\dot{\theta}$.

If the radar system is not working, the pilot makes use of earth-based tracking radar to evaluate the approach trajectory. If nominal conditions prevail, the pilot directs thrust 28° above the lunar horizon for 217 seconds at which time the vehicle is rotated so that thrust is directed about 19° above the horizon. The second thrust angle is maintained for 69 seconds at which time thrust is terminated. If nominal conditions do not prevail, the pilot must in essence formulate a new nominal by obtaining new values for the thrusting angles and the thrusting time (from tabulated data, for example).

Hodograph Target Area

The precise accomplishment of the piloting task consisted of achieving $\dot{R} = 0$ feet per second, $R\dot{\theta} = 5285$ feet per second, and $h = 486\ 000$ feet. Such a degree of accuracy would be extremely difficult to attain. Thus, it was decided that the pilot should strive to attain these conditions, but that any combination of conditions that resulted in an orbit having a maximum apocynthion altitude of about 611 000 feet and a minimum pericynthion altitude of about 361 000 feet would be acceptable. In addition, there should be no difficulty in obtaining the nominal altitude ±10 000 feet at thrust termination. Thus, a hodograph target area was specified by computing (see ref. 2) the range of velocities that would be acceptable at each of these altitudes with the specified apocynthion and pericynthion boundaries (fig. 8). For a given altitude at thrust termination, the limit on apocynthion altitude specifies an elliptical boundary and the limit on pericynthion altitude specifies a hyperbolic boundary. The area common to both sets of closed curves specifies the combinations of $\dot{R}$ and $R\dot{\theta}$ that must be attained within 10 000 feet of the nominal altitude at thrust termination in order that the established orbit possess a minimum altitude at pericynthion of 361 000 feet and a maximum altitude at apocynthion of 611 000 feet.

RESULTS AND DISCUSSION

The results of the investigation obtained with the authors as pilots are divided into two sections: the first deals with the situation involving radar information; and the second, with the situation involving earth-based tracking information. Each section is divided into two subsections concerning the nominal trajectory and the off-nominal trajectories.

Nominal Trajectory

The problem was initiated at an altitude of about 601 600 feet with the vehicle alined with the local horizontal. This condition resulted in the longitudinal axis of the vehicle being directed about 25° above the line of sight to the lunar horizon. The pilot then had 20 seconds to rotate the vehicle to the desired attitude $K = 28^{\circ}$ before thrust application at 580 000 feet. Results of a typical piloted nominal trajectory are shown in figure 9. The solid line on the hodograph is the nominal path and the dashed line is the result of the piloted flight. The results show that the piloted trace followed the nominal path very closely throughout most of the flight and terminated with approximately the conditions necessary for the establishment of a 486 000-foot-altitude circular orbit.

A summary of terminal conditions obtained by flying the nominal trajectory is presented in figure 10. The terminal values of the circumferential and radial velocity components were well within the hodograph target area and were in general within ±10 feet per second of the desired values. The final altitude was generally higher than the nominal but by less than 5000 feet. The characteristic velocity required by the pilots was about 2 percent less than the results from the digital computations of the nominal path. However, it was determined that this difference approached the accuracy of the analog computations. The pilots experienced no difficulty in maintaining the proper vehicle orientation along the trajectory or in attaining the desired terminal conditions. In addition, there was no difference (fig. 10) between the results of flights made by using the scope presentation for attitude reference and those using the K-meter and 3-axis ball indicator. Thus rather crude measurement of the thrust angle appears to be acceptable for performing the maneuver.

Off-Nominal Trajectories

A total of seven off-nominal combinations of initial conditions were used in the investigation. The flight history of a typical off-nominal flight is presented in figure 11. The initial conditions for this flight were a radial velocity of -1188 feet per second, a circumferential velocity of 8075 feet per second, and an altitude of 591 600 feet. At flight initiation, the pilot pitched the vehicle to thrust more in the radial direction than was nominally the case in an attempt to minimize the effect of the off-nominal conditions on altitude and rate of descent. In addition, thrust was initiated approximately 6 seconds later than nominally to account for the initial low value of circumferential velocity. Approximately 94 seconds after thrust initiation, the pilot observed that his present attitude would not result in the reduction of the rate of descent to zero at the desired altitude. The pilot therefore pitched to a nearly vertical attitude to reduce the rate of descent more rapidly. This maneuver was insufficient to keep the vehicle from descending below 486 000 feet. Therefore, the pilot introduced a small rate of climb which was maintained until the vehicle climbed back to 486 000 feet. The pilot then pitched down so that thrust was directed about 19° above the horizon and performed a constant-altitude deceleration maneuver until the proper circumferential velocity was attained.

Approximately 16 flights were made for each of the seven off-nominal trajectories used in this study. The terminal conditions obtained by flying the off-nominal trajectories are summarized in figure 12. The terminal values of the radial and out-of-plane velocity components were generally within ±10 feet per second whereas the more critical circumferential velocity component was generally within ±5 feet per second of the desired value. The final altitude was generally within ±5000 feet of that desired and the out-of-plane displacement was generally less than 1000 feet. The characteristic velocity required to fly the off-nominal trajectories was always within 5 percent of that required for a perfectly flown nominal trajectory.

Supplementary Investigation

An additional situation was considered in which it was assumed that the only information available to the pilot was obtained from an earth-based tracking system. The pilot uses earth-based tracking information to evaluate the approach trajectory. If nominal conditions prevail, the pilot maintains the nominal thrust angles and terminates thrust on time. If nominal conditions do not prevail, the pilot obtains new values for the thrusting angles and thrusting time.

For the nominal trajectory used in this investigation, thrust is directed 28° above the lunar horizon for 217 seconds followed by thrusting 19° above the horizon for 69 seconds. A summary of the terminal conditions is presented in figure 13. The results show that the end conditions were of the same order of magnitude as for the flights using velocity and altitude information. Thus, if the vehicle is on the nominal lunar approach trajectory, the pilot can establish very nearly circular orbits at the proper altitude.

However, the actual velocity and altitude may differ from that indicated by earth-based tracking and the pilot may not actually be on a nominal approach trajectory. The results of flights made by using the nominal piloting procedure under the influence of errors typical of earth-based tracking information are summarized in figure 14. The initial errors considered in this investigation result in rather significant errors in velocity and altitude at thrust termination. However, the altitude at pericynthion of the resulting lunar orbits exceeded 150 000 feet.

CONCLUDING REMARKS

A fixed-base simulator study has been made of the ability of pilots to modify lunar approach trajectories to establish circular orbits about the moon at an altitude of 486 000 feet by using a simplified guidance scheme. The study permitted all six rigid-body degrees of freedom of the vehicle. It was assumed that the vehicle had no automatic damping or control. The pilot's task was to maintain a constant angle between the thrust vector and the line of sight to the lunar horizon until reaching the proper altitude with zero rate of descent. The pilot then performed a constant-altitude deceleration maneuver to attain circular velocity.

The results of the investigation showed that the pilots could consistently establish orbits lying well within an altitude range from 361 000 to 611 000 feet by using rather crude thrust angle measurements. The fuel required corresponded to a characteristic velocity that was within 5 percent of that computed for a perfectly flown trajectory.

The pilots could consistently establish circular orbits from the nominal-approach trajectory in the absence of velocity and altitude information. This condition was accomplished by maintaining a constant thrust angle with respect to the lunar horizon for a given time followed by a second constant thrust angle for a second specified time. This procedure did not result in the establishment of near-circular orbits when used under the influence of possible earth-based tracking errors but did result in establishing nonimpacting orbits.

Langley Research Center,
National Aeronautics and Space Administration,
Langley Station, Hampton, Va., November 5, 1964.

APPENDIX

EQUATIONS OF MOTION

The equations of motion used in this investigation are presented in this appendix. The force equations are written with respect to an inertial axis system located at the center of a nonrotating spherical moon and the moment equations were written about the vehicle body axes as follows:

$$\ddot{x} = \frac{F}{m}(\cos\psi \sin\theta \cos\phi + \sin\psi \sin\phi) - g_m\left(\frac{R_m}{R}\right)^2 \frac{x}{R}$$

$$\ddot{y} = \frac{F}{m}(\sin\psi \sin\theta \cos\phi - \cos\psi \sin\phi) - g_m\left(\frac{R_m}{R}\right)^2 \frac{y}{R}$$

$$\ddot{z} = \frac{F}{m} \cos\theta \cos\phi - g_m\left(\frac{R_m}{R}\right)^2 \frac{z}{R}$$

$$\dot{p} = \frac{1}{I_X}\left[M_X - qr\left(I_Z - I_Y\right)\right]$$

$$\dot{q} = \frac{1}{I_Y}\left[M_Y - pr\left(I_X - I_Z\right)\right]$$

$$\dot{r} = \frac{1}{I_Z}\left[M_Z - pq\left(I_Y - I_X\right)\right]$$

In addition, several auxiliary equations were used:

$$\dot{\psi} = \frac{r \cos\phi + q \sin\phi}{\cos\theta}$$

APPENDIX

$$\dot{\theta} = q \cos \phi - r \sin \phi$$

$$\dot{\phi} = p + \dot{\psi} \sin \theta$$

$$M_X = \frac{\partial M_X}{\partial \delta_X} \delta_X$$

$$M_Y = \frac{\partial M_Y}{\partial \delta_Y} \delta_Y$$

$$M_Z = \frac{\partial M_Z}{\partial \delta_Z} \delta_Z$$

$$m = m_o + \int_0^t \dot{m} \, dt$$

$$\dot{m} = \frac{F}{g_e I_{sp}}$$

$$F = - \frac{\partial F}{\partial \delta_F} \delta_F$$

$$R = R_m + h = \sqrt{x^2 + y^2 + z^2}$$

$$\dot{R} = \frac{x\dot{x} + y\dot{y} + z\dot{z}}{R}$$

$$R\dot{\theta} = \sqrt{\dot{x}^2 + \dot{y}^2 + \dot{z}^2 - \dot{R}^2}$$

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1. Houbolt, John C.; Bird, John D.; and Queijo, Manuel J.: Guidance and Navigation Aspects of Space Rendezvous. Proceedings of the NASA-University Conference on the Science and Technology of Space Exploration, Vol. 1, NASA SP-11, 1962, pp. 353-366. (Also available as NASA SP-17.)

2. Queijo, M. J.; and Riley, Donald R.: A Fixed-Base-Simulator Study of the Ability of a Pilot To Establish Close Orbits Around the Moon. NASA TN D-917, 1961.

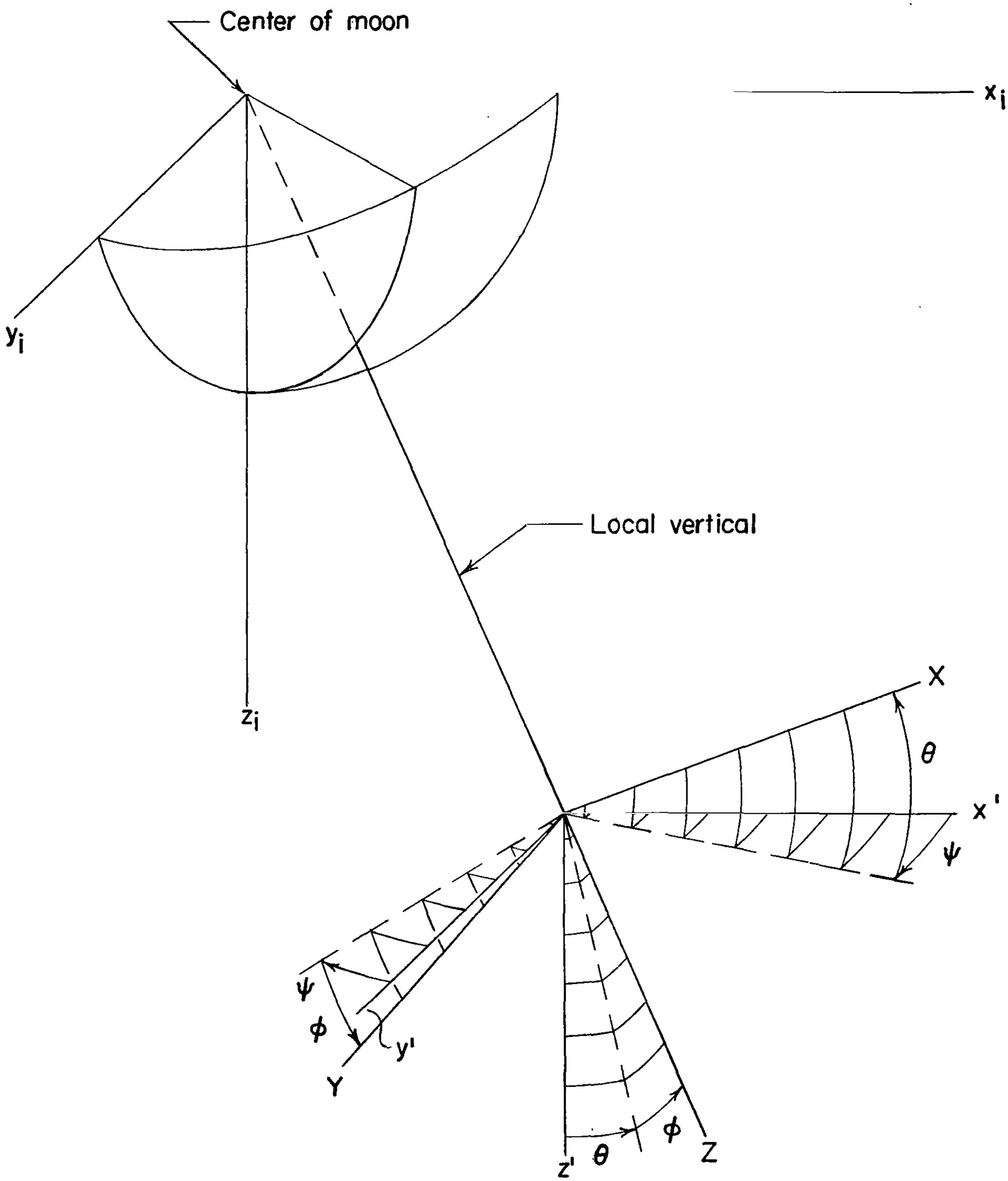

Figure 1.- Inertial-axis and vehicle-axis systems.

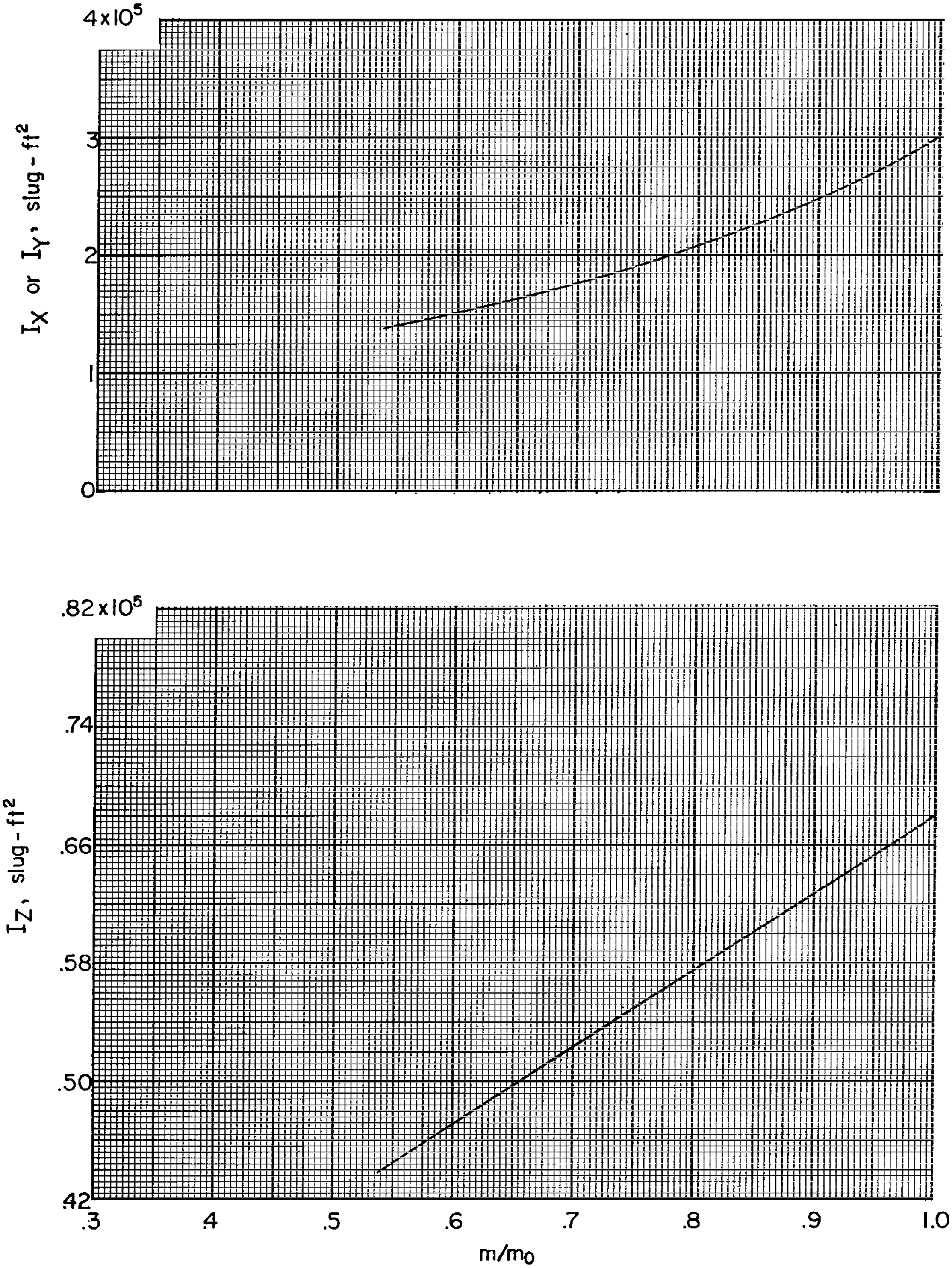

Figure 2.- Vehicle characteristics.

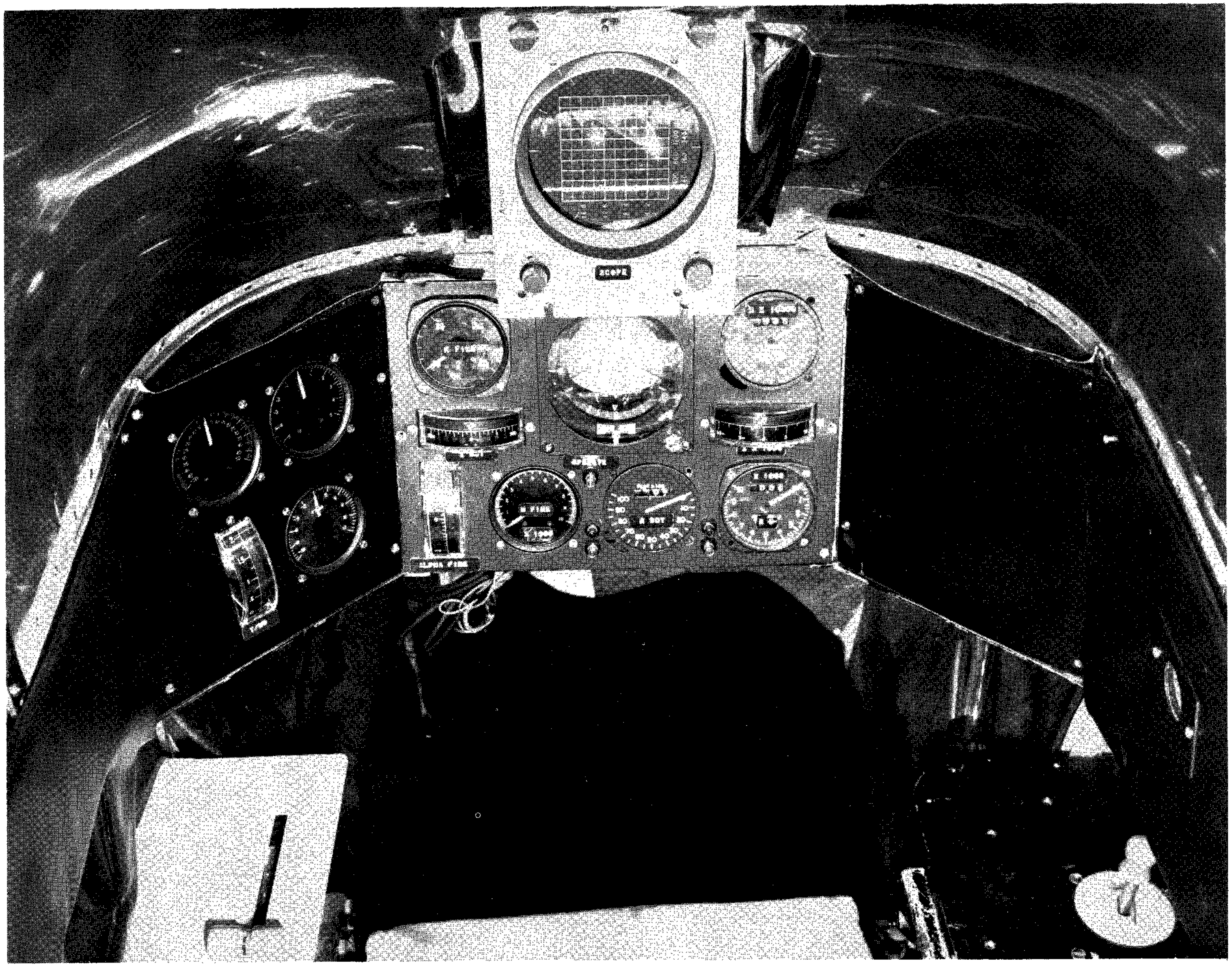

Figure 3.- General layout of cockpit.

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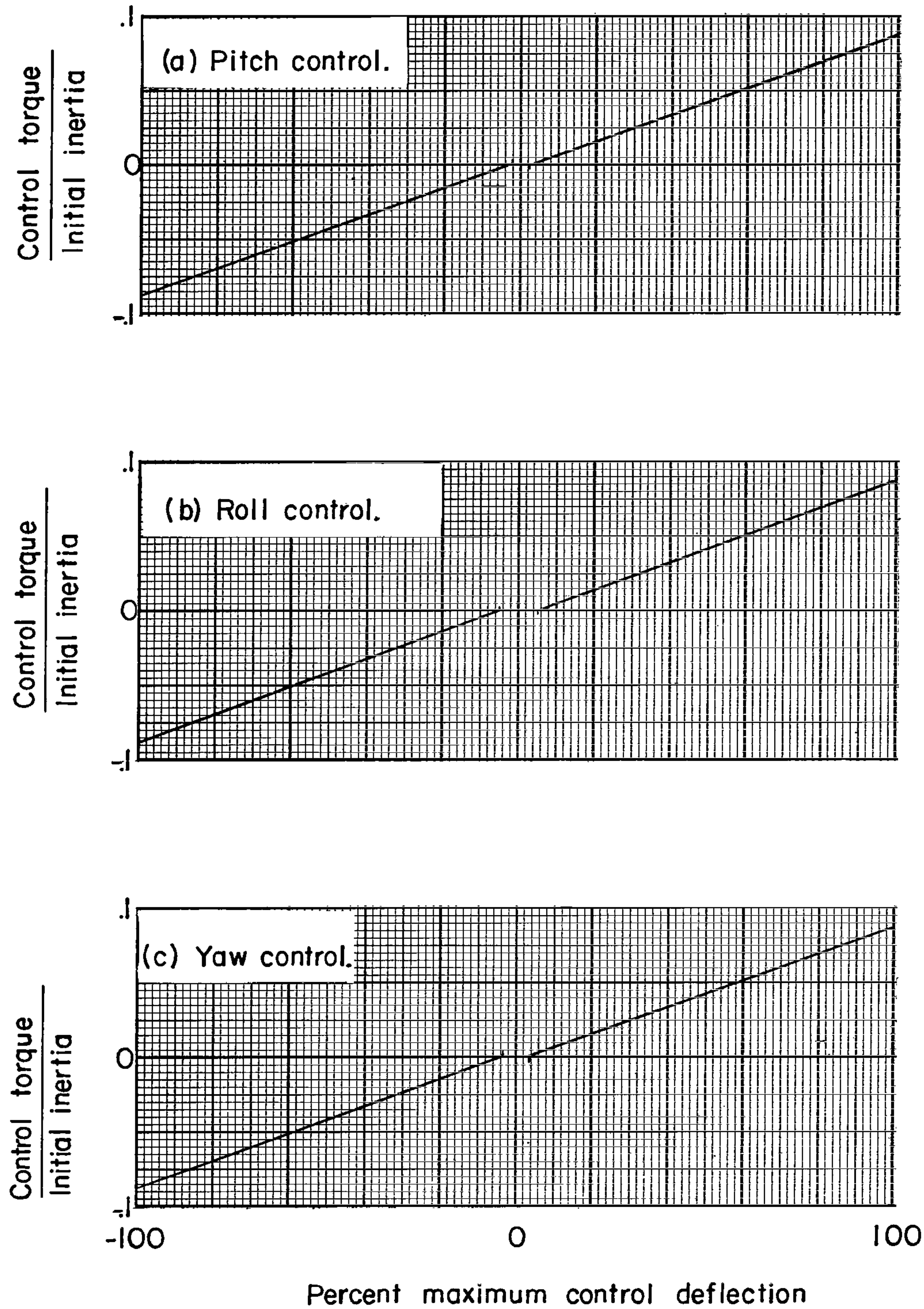


Figure 4.- Initial variations of control torque with controller deflection.

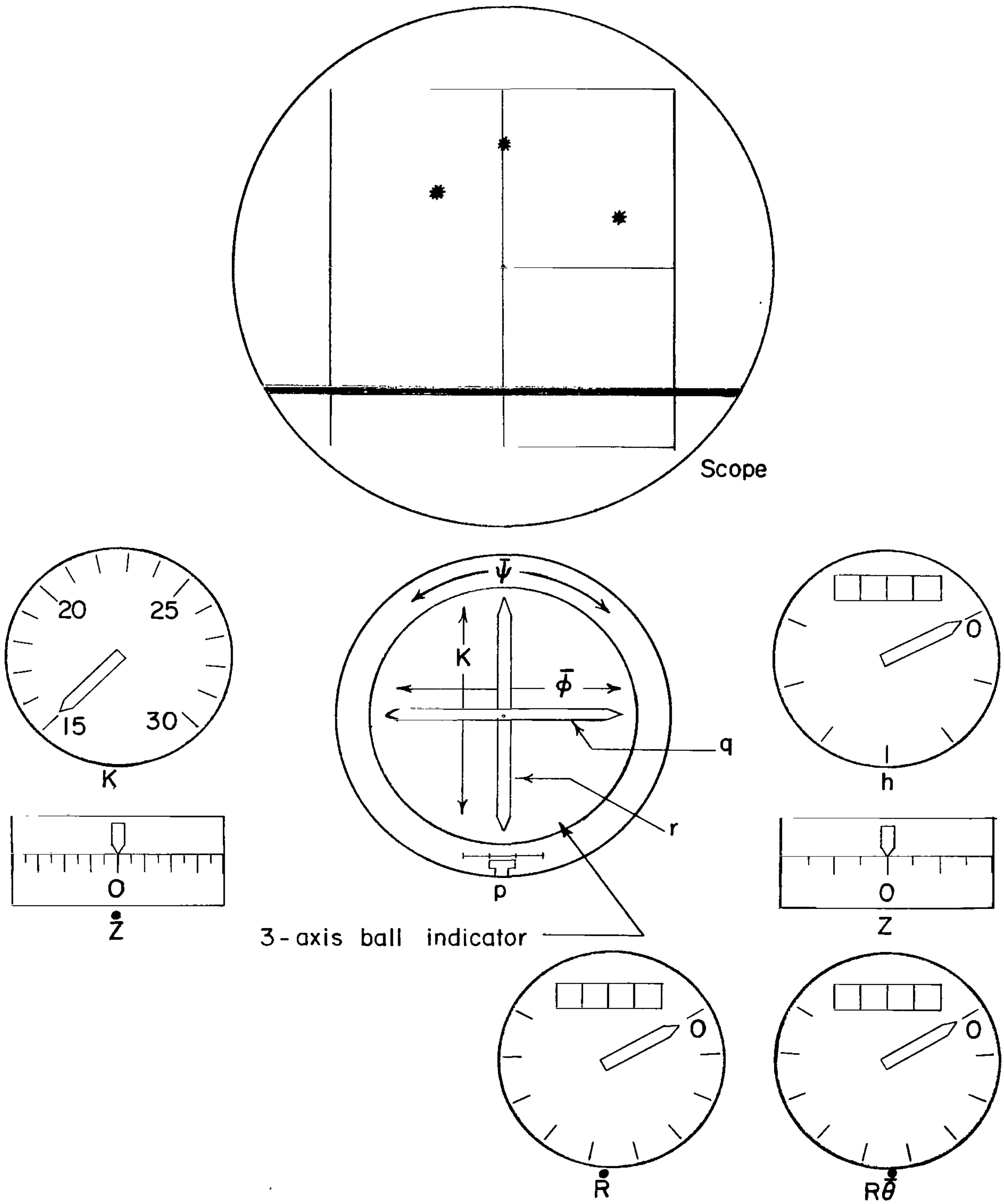


Figure 5.- Instrument panel.

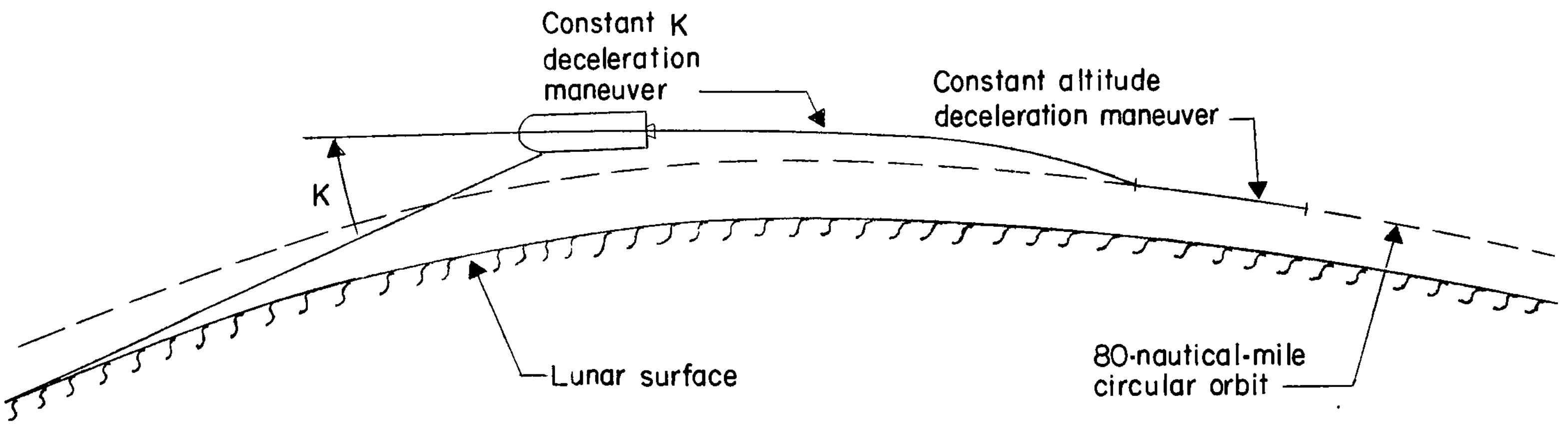


Figure 6.- Nominal injection maneuver.

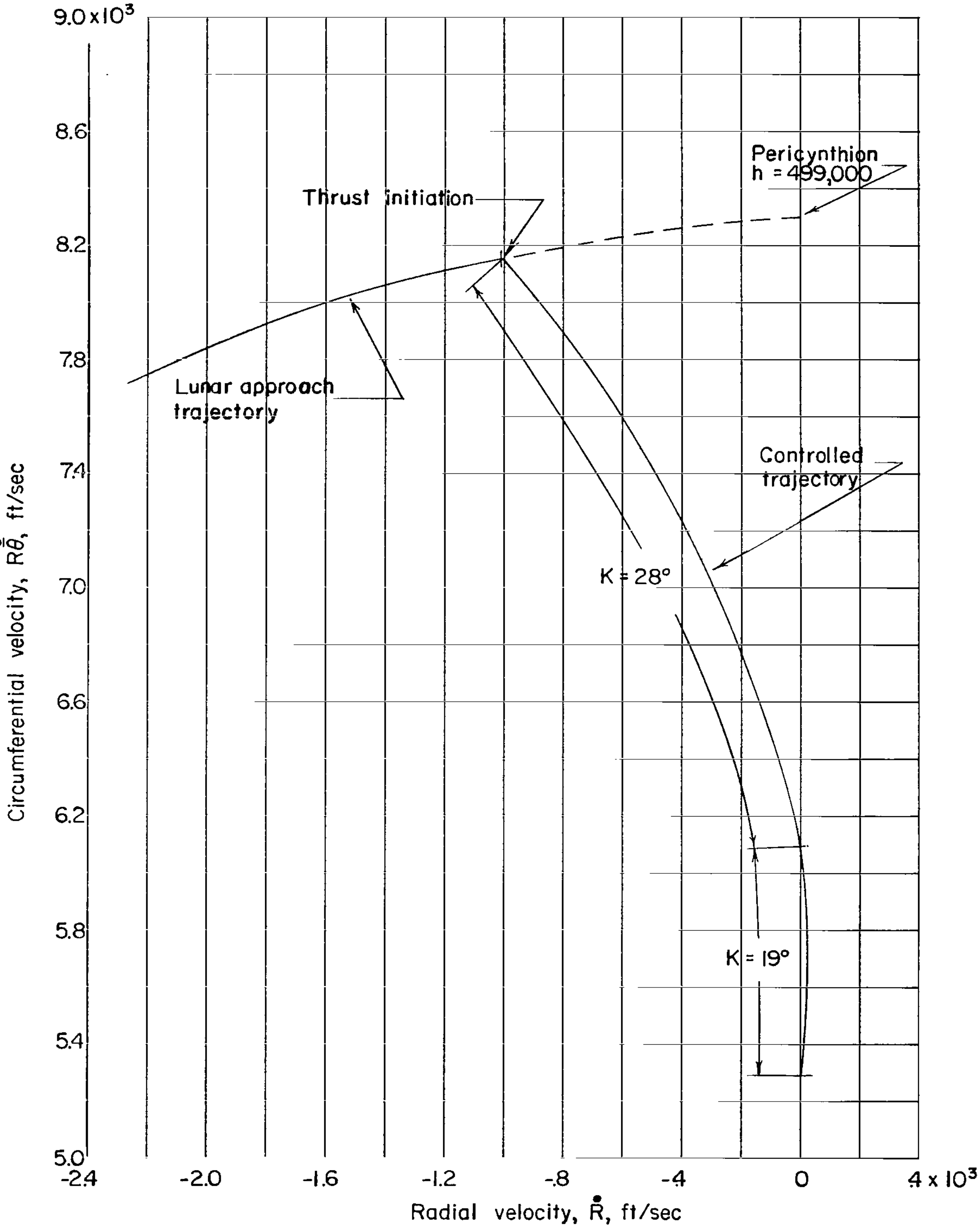


Figure 7.- Nominal trajectory in hodograph plane.

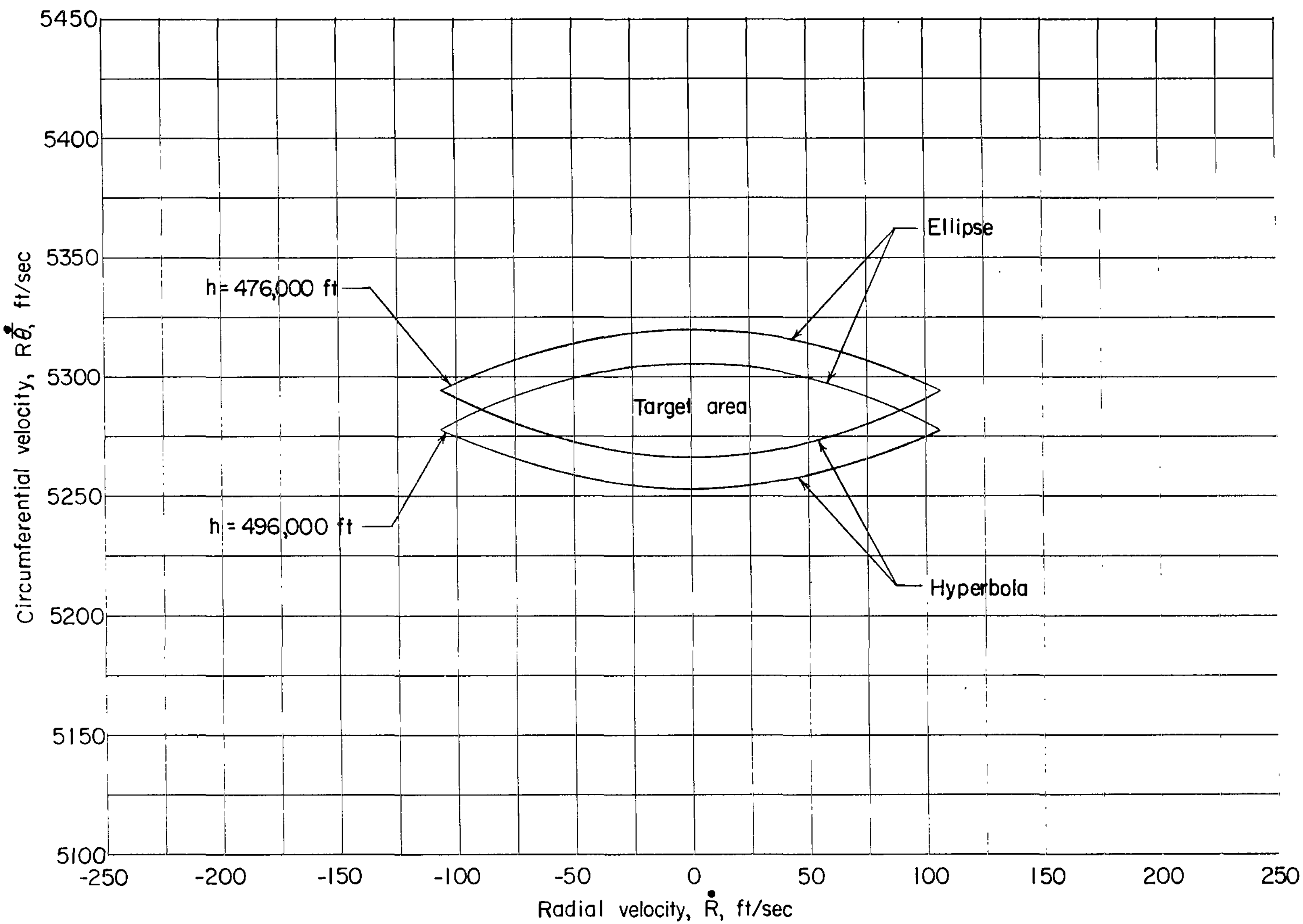
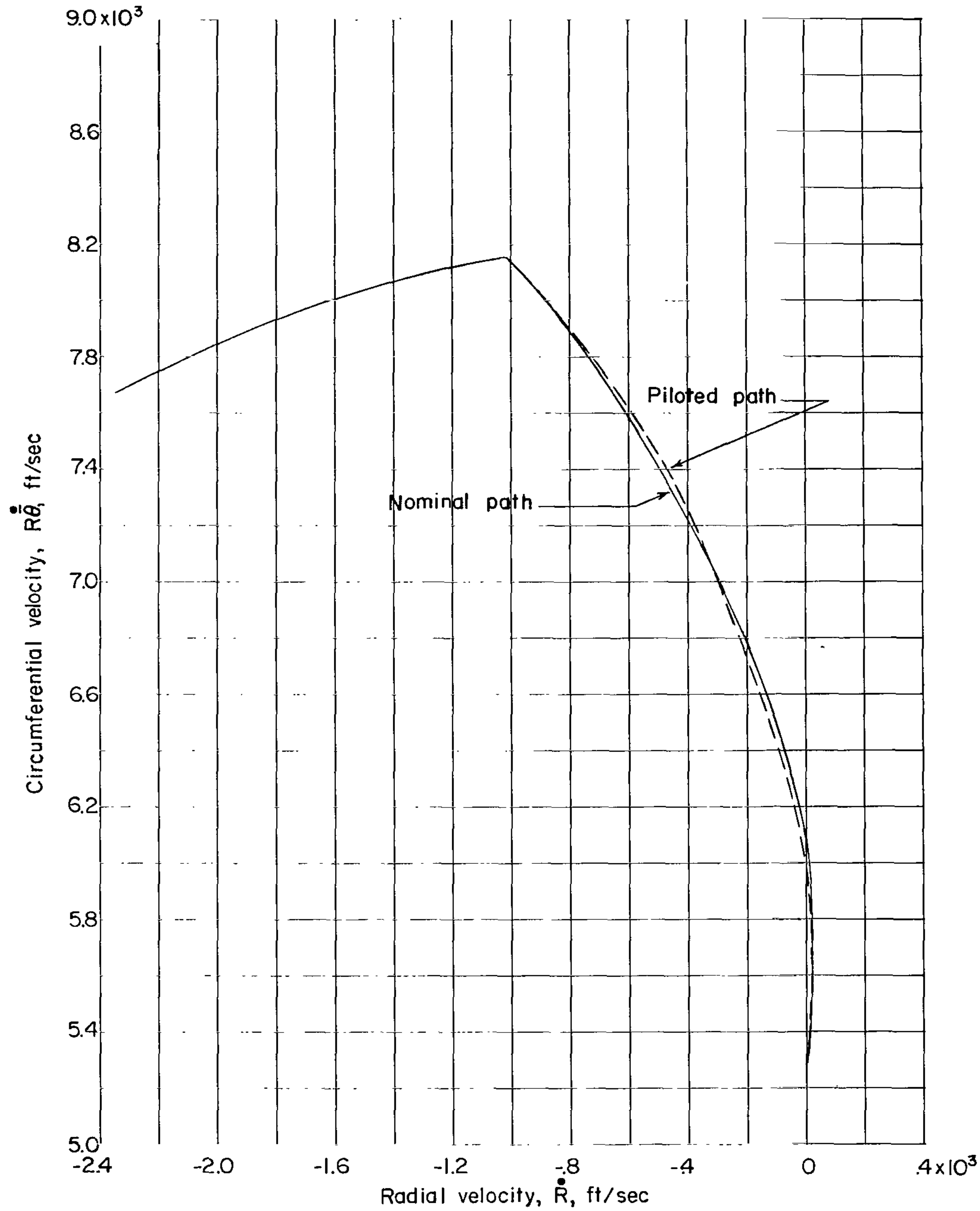
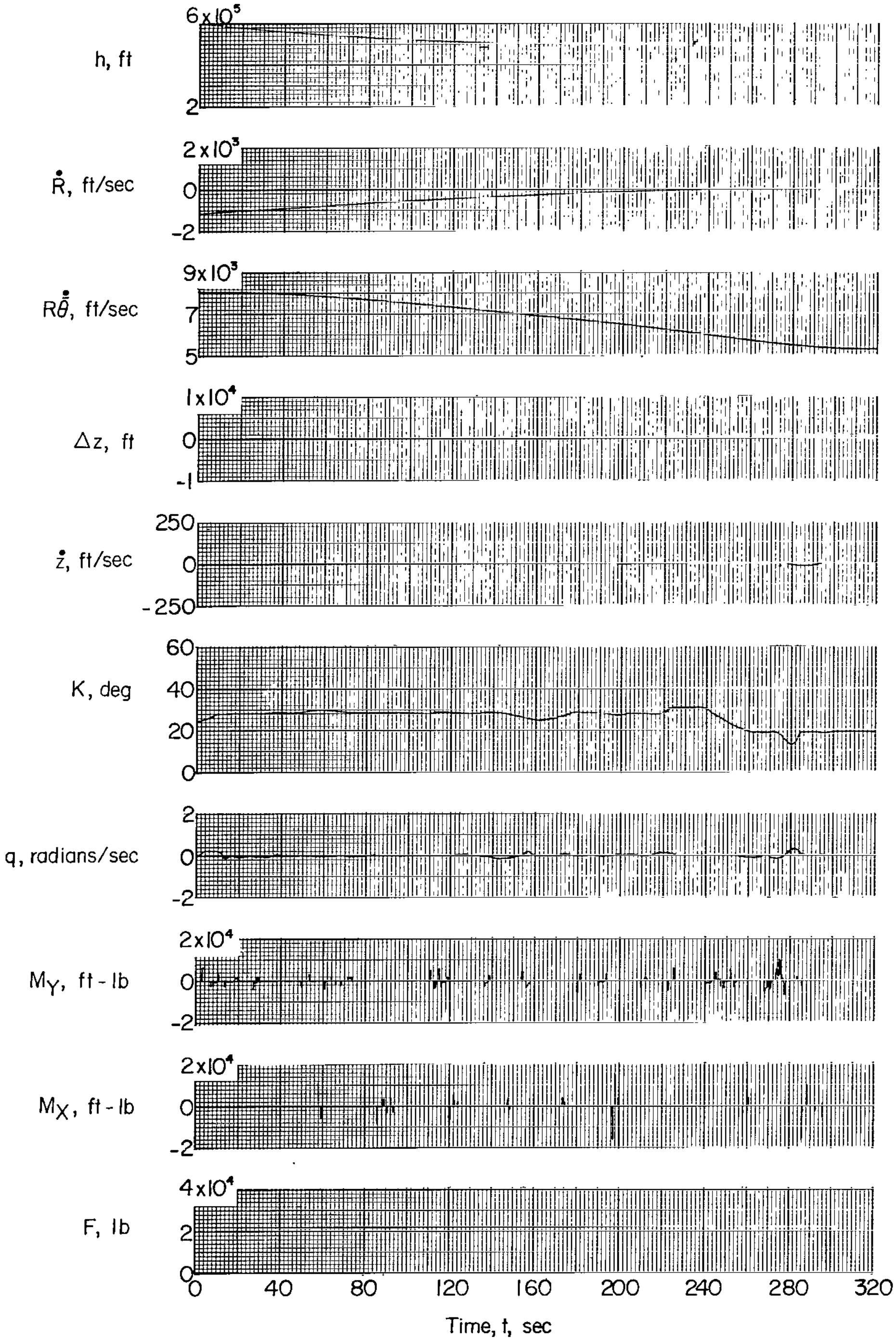


Figure 8.- Hodograph target area.



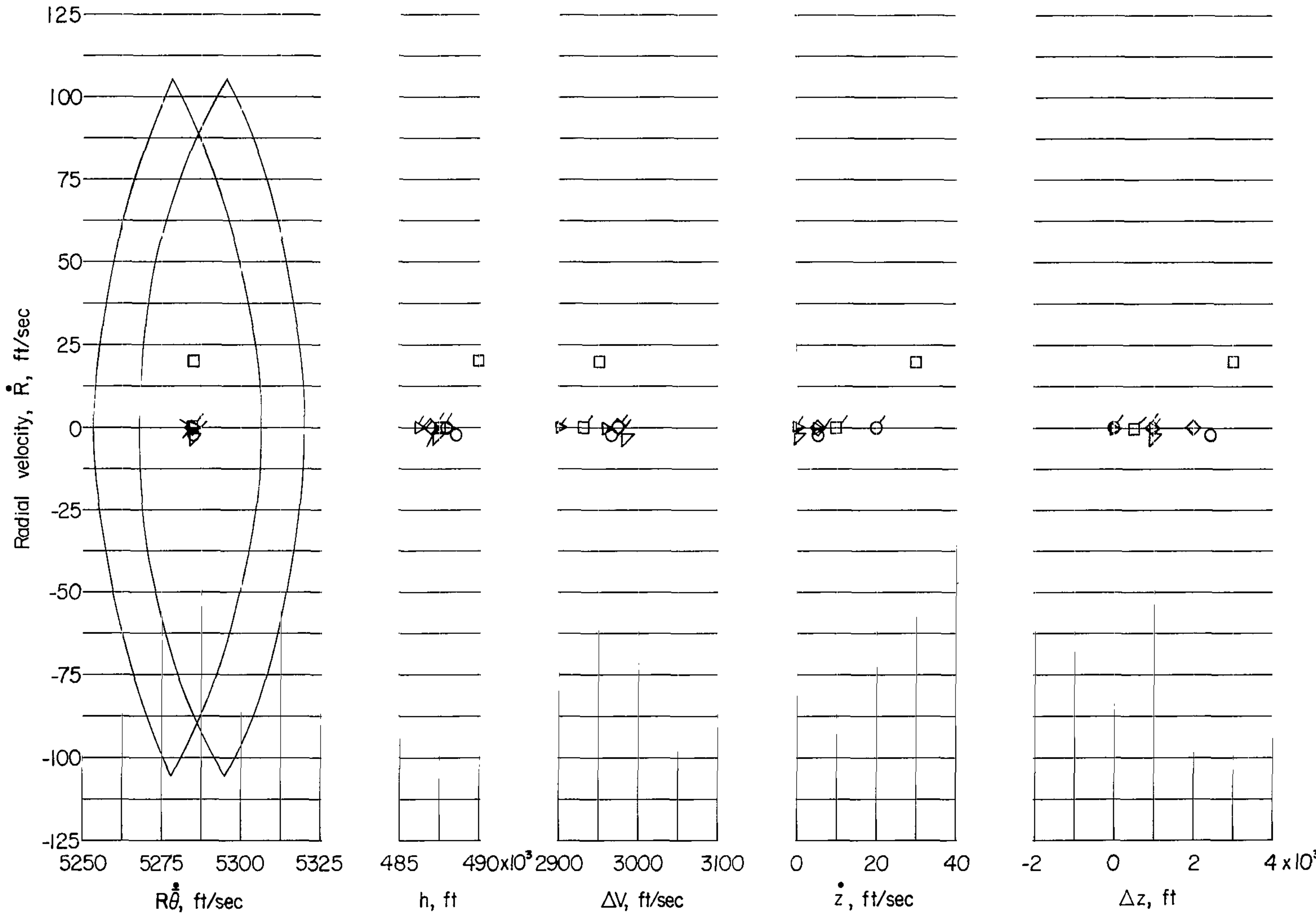
(a) Hodograph.

Figure 9.- Typical flight history of nominal trajectory. $\dot{R}_o = -1138$ ft/sec; $\left(R\dot{\theta}\right)_o = 8125$ ft/sec; $h_o = 601\ 600$ ft.



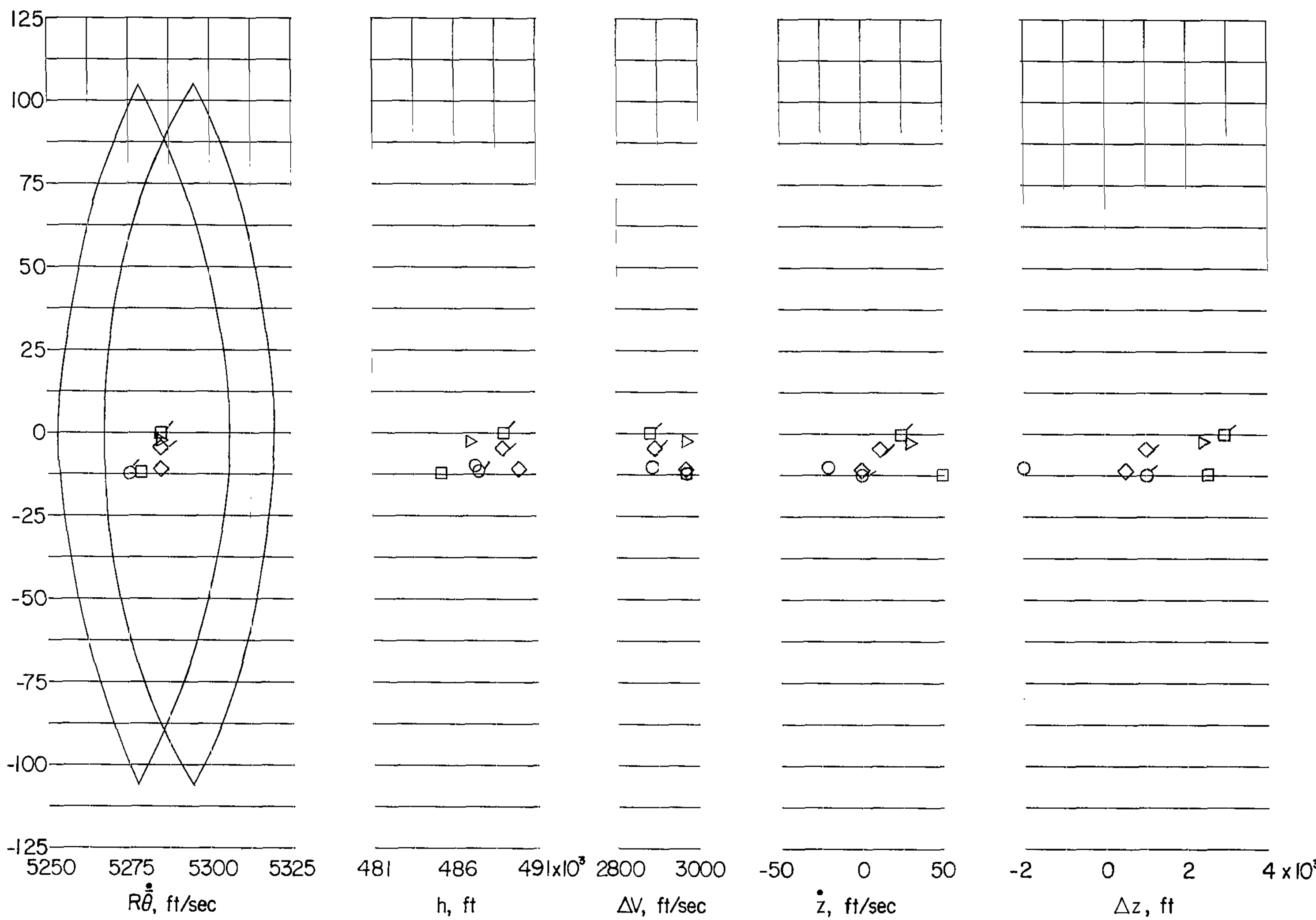
(b) Time histories.

Figure 9.- Concluded.



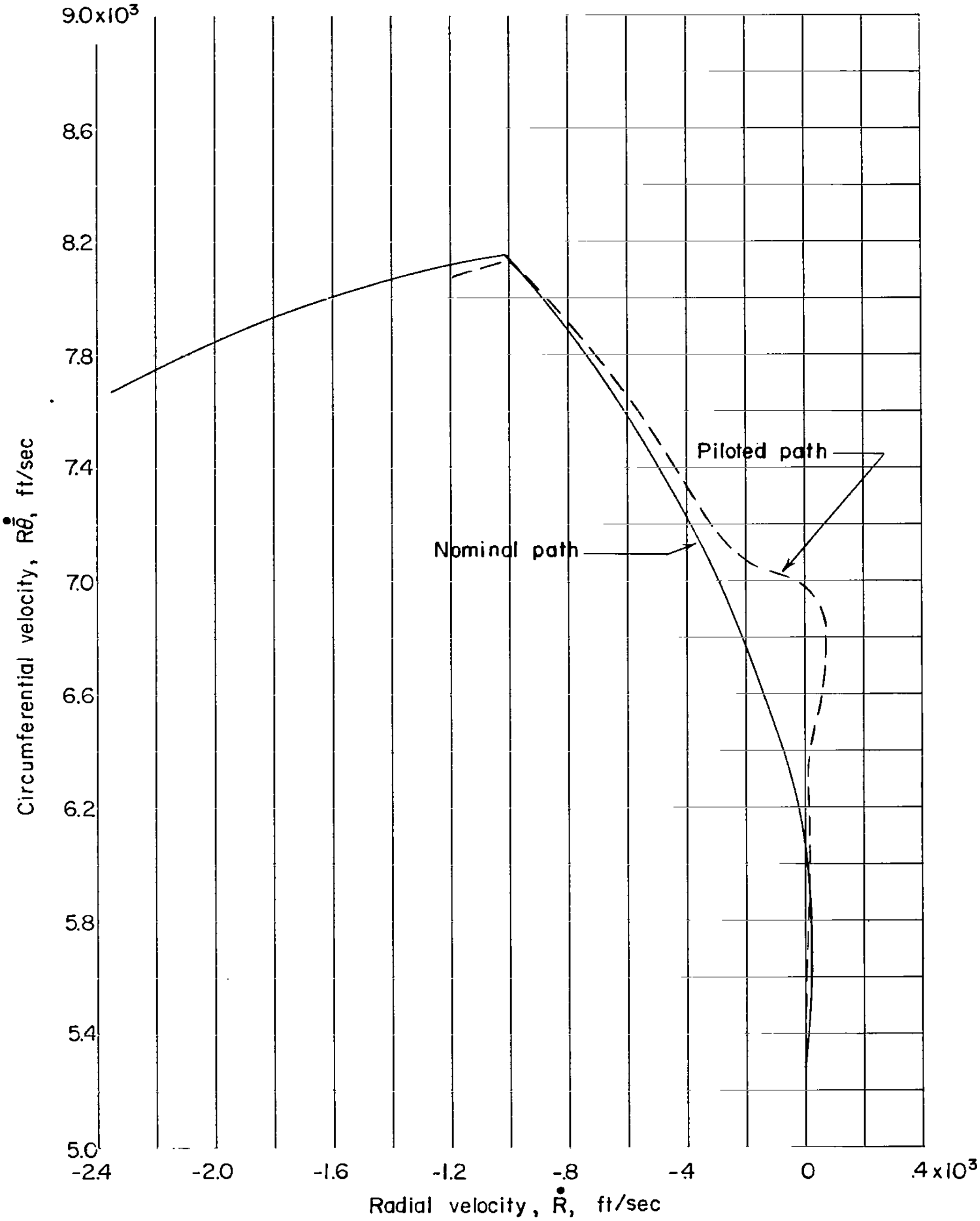
(a) Pilot 1.

Figure 10.- Terminal conditions of nominal flights. (Flagged symbols denote flights using scope presentation for attitude reference. Different symbols differentiate between runs.)



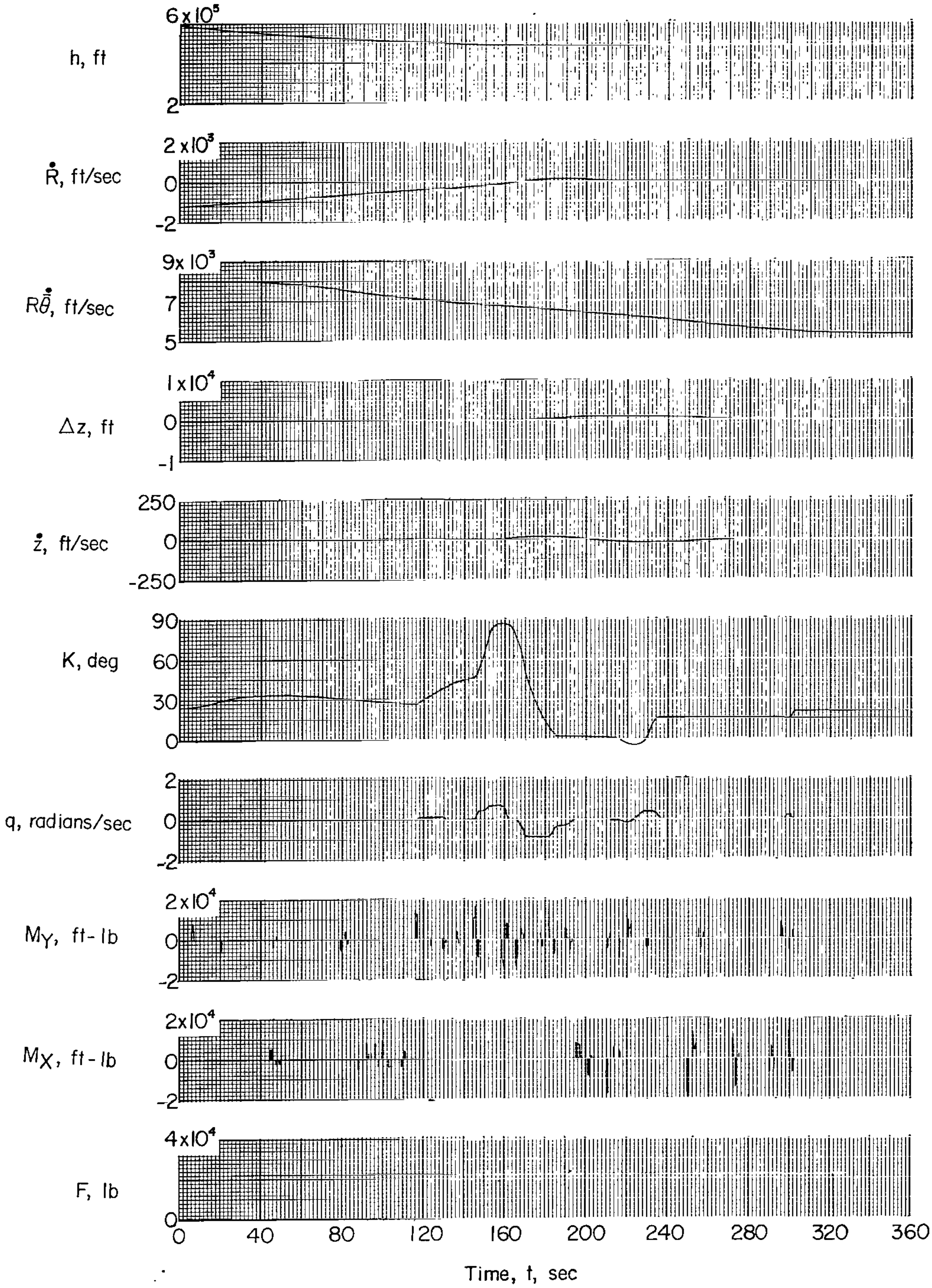
(b) Pilot 2.

Figure 10.- Concluded.



(a) Hodograph.

Figure 11.- Typical flight history of off-nominal trajectory. $\dot{R}_o$ = -1188 ft/sec; $\left(R\dot{\theta}\right)_o$ = 8075 ft/sec; h_o = 591 600 ft.



(b) Time histories.

Figure 11.- Concluded.

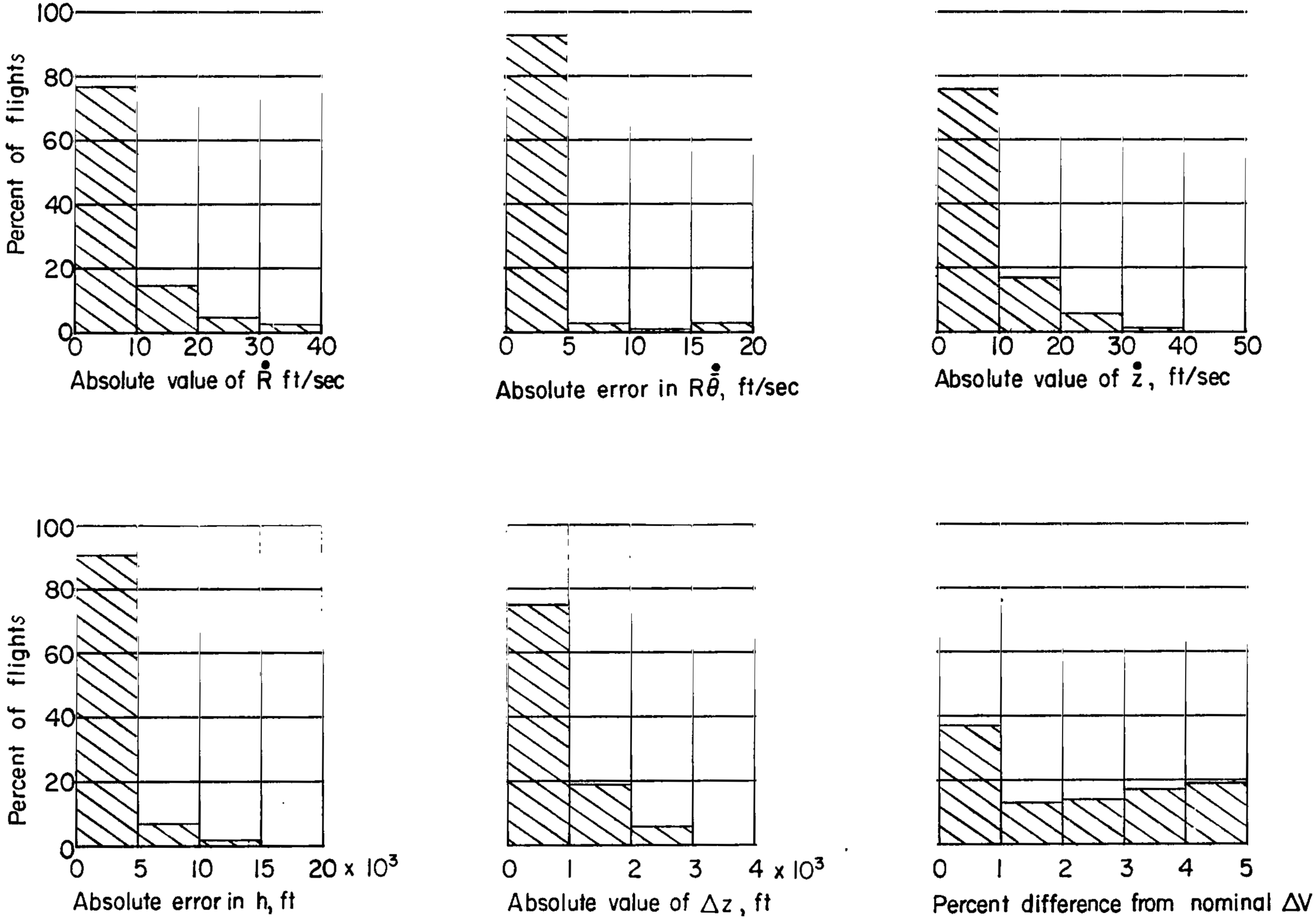


Figure 12.- Summary of terminal conditions of off-nominal flights.

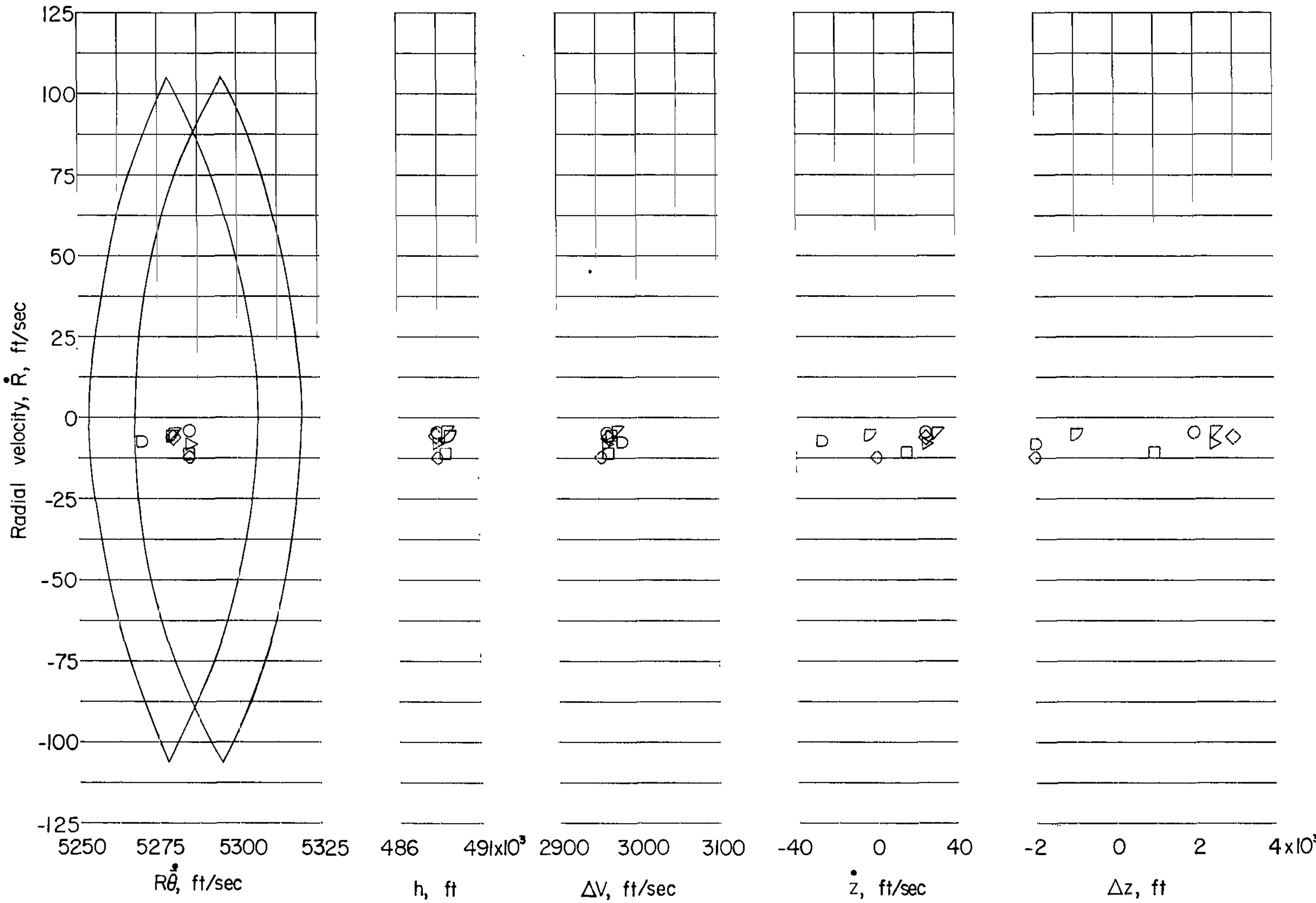
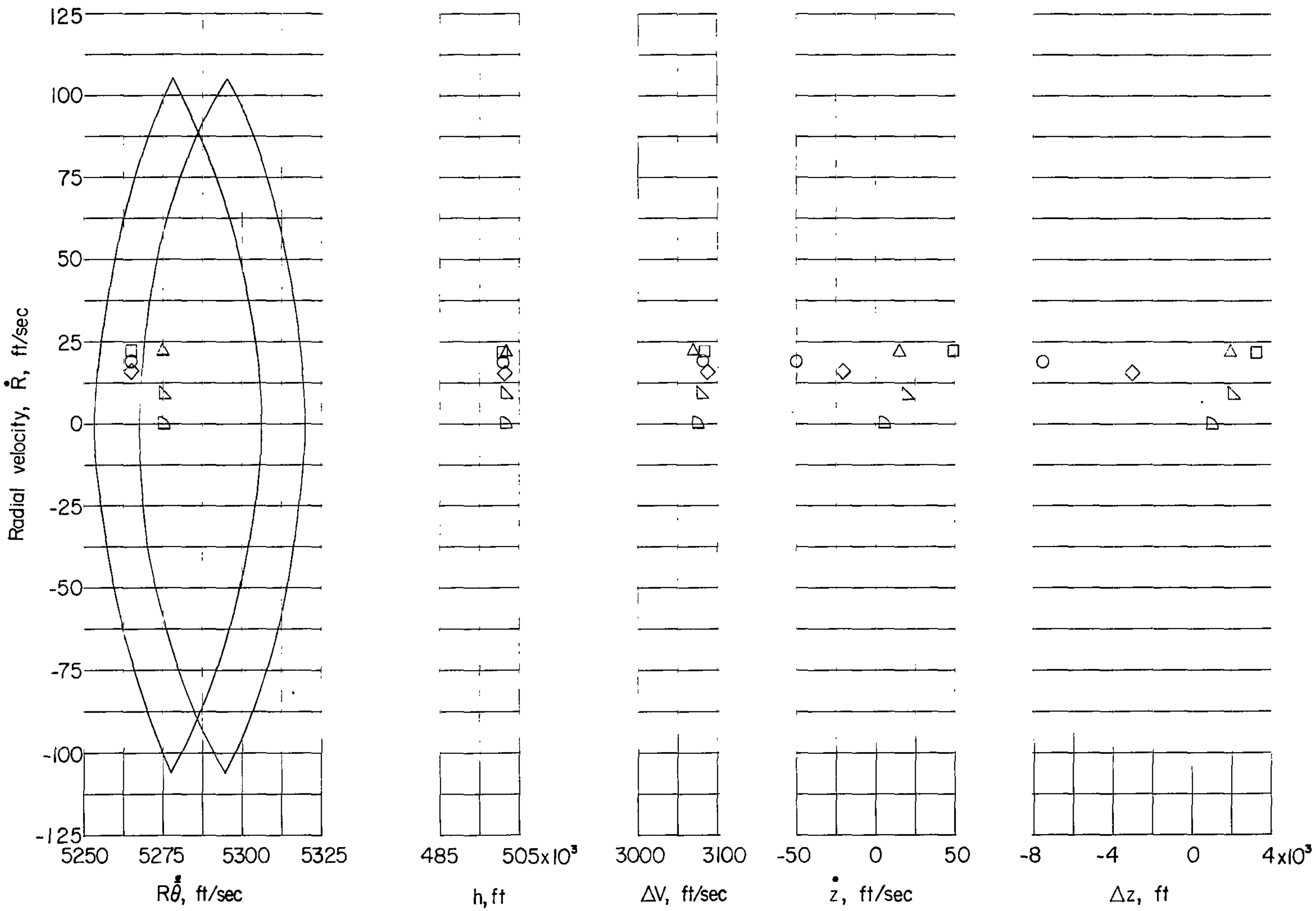
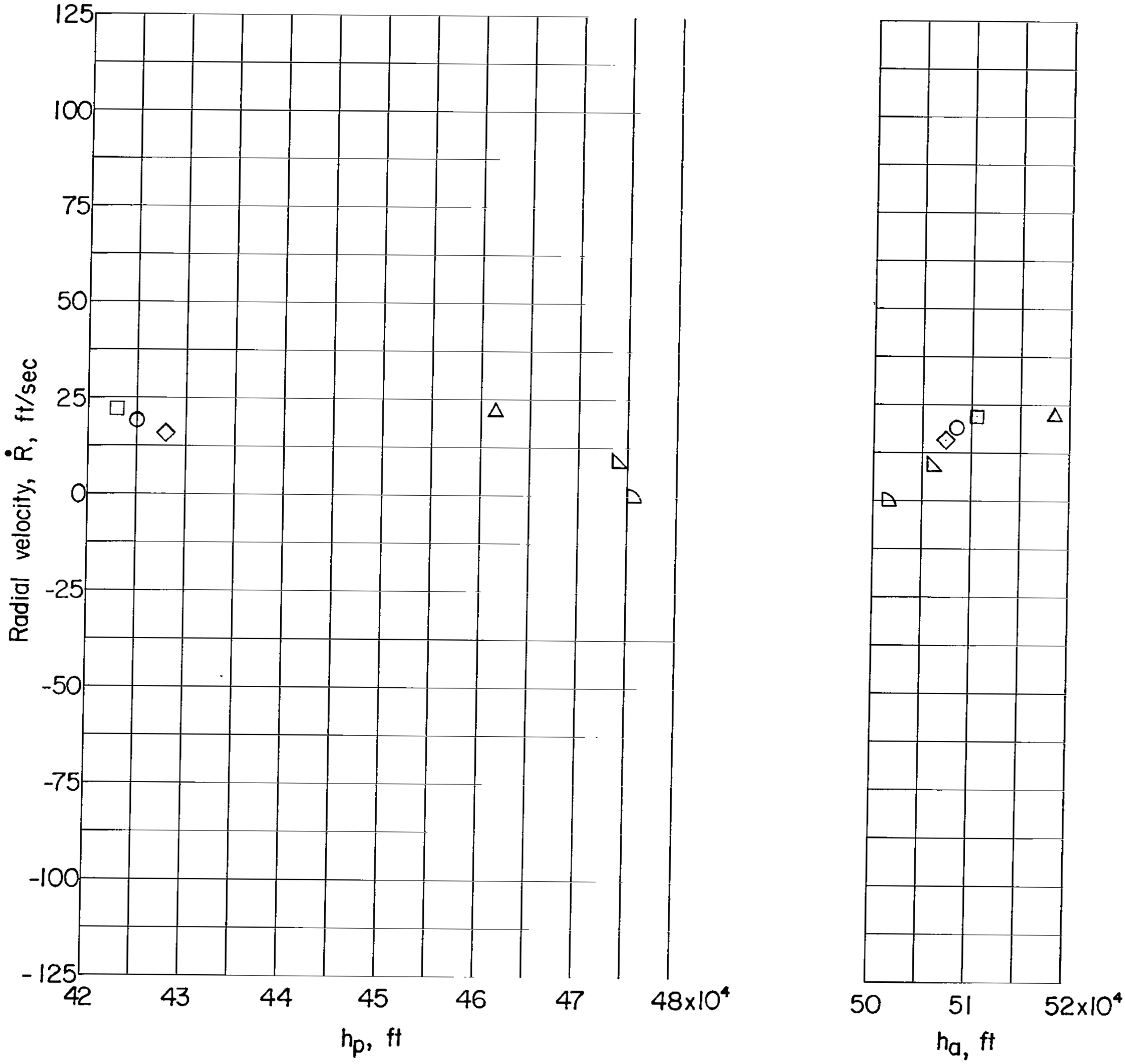


Figure 13.- Terminal conditions of nominal flights, radar inoperative. (Different symbols differentiate between runs.)



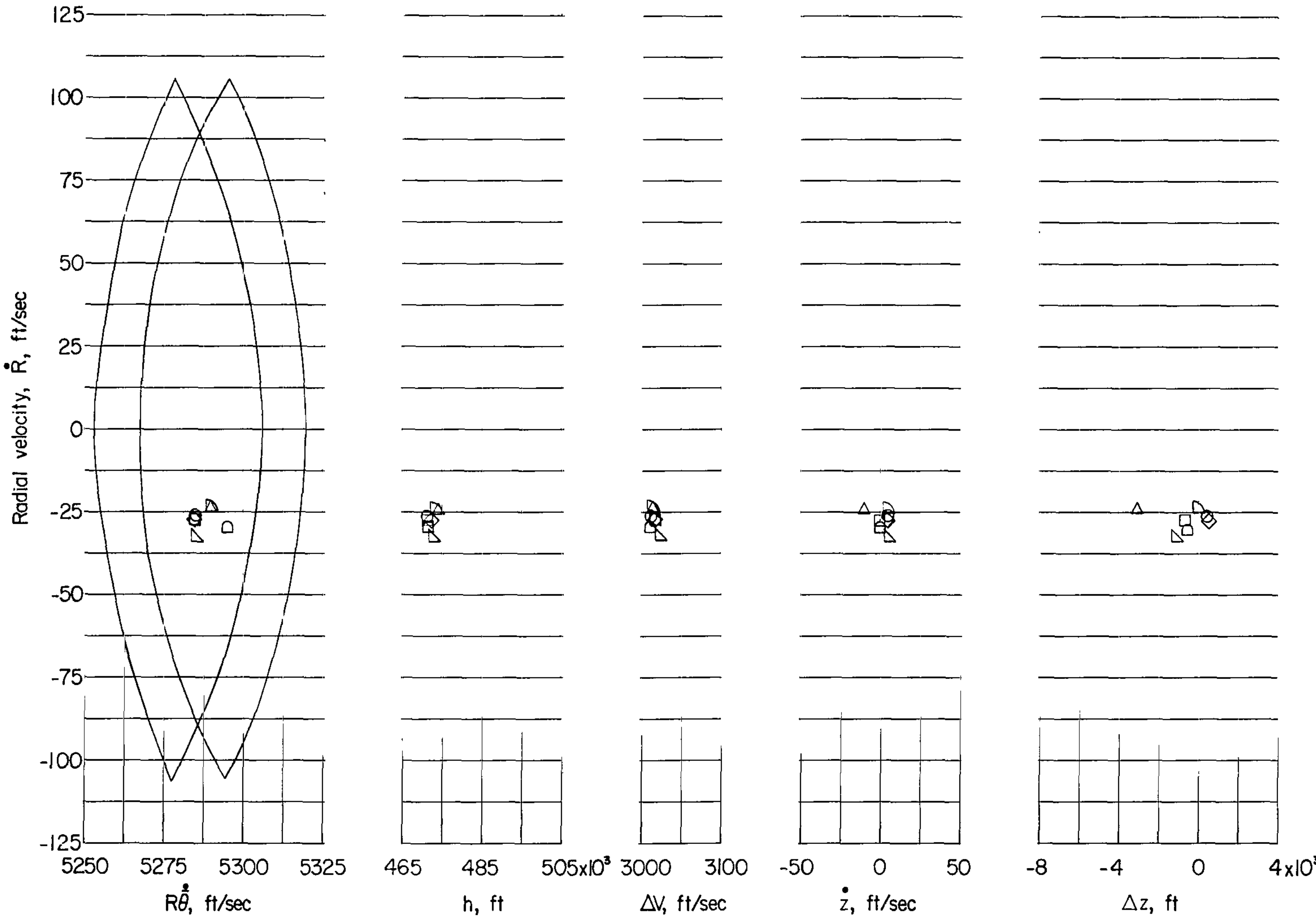
(a) $\dot{R}_o$ = -1088 ft/sec; $\left(R\dot{\theta}\right)_o$ = 8125 ft/sec; h_o = 601 600 ft.

Figure 14.- Terminal conditions of flights under influence of possible earth-based tracking errors, radar inoperative. (Different symbols differentiate between runs.)



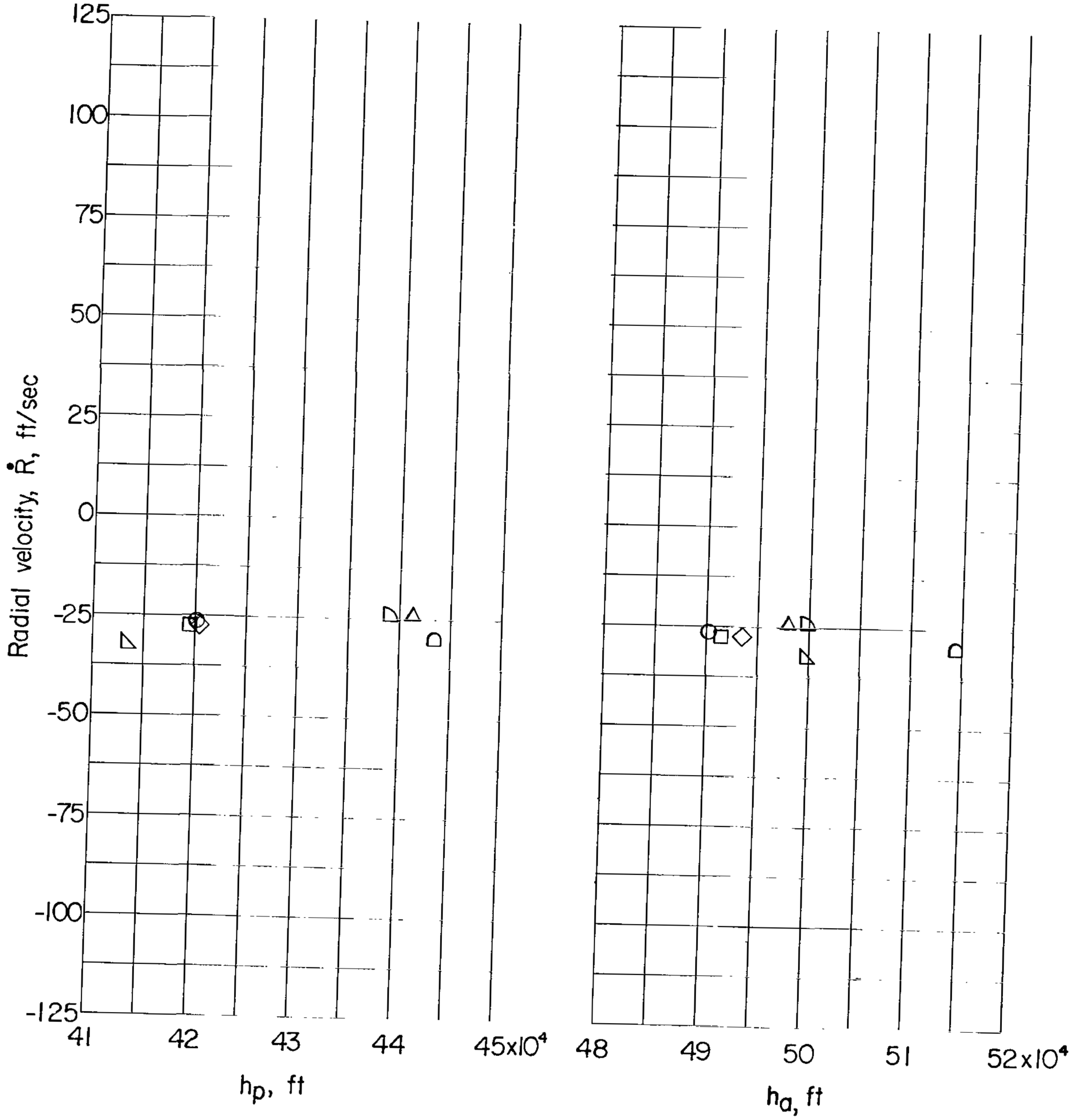
(a) Concluded.

Figure 14.- Continued.



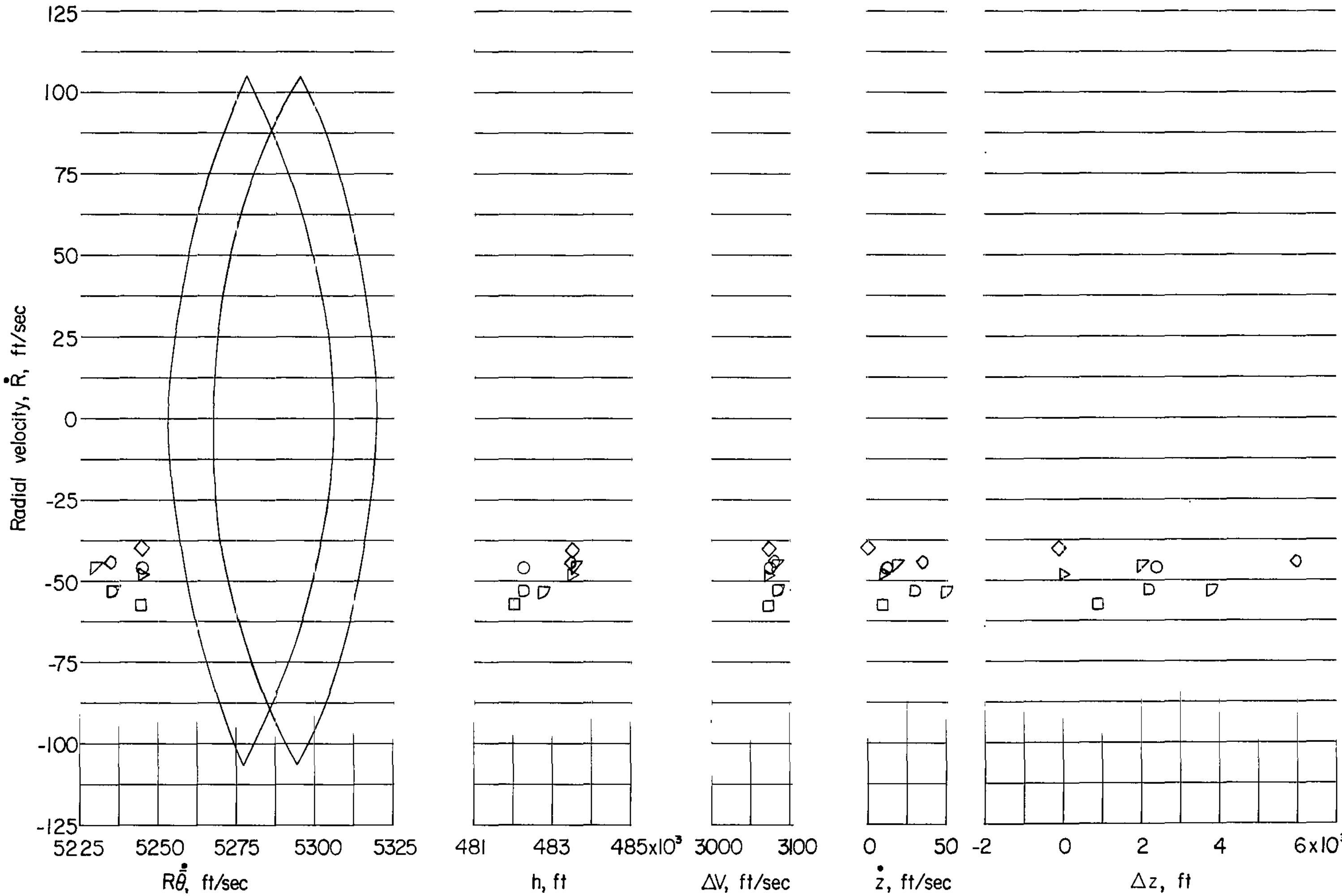
(b) $\dot{R}_o = -1188$ ft/sec; $\left(R\dot{\theta}\right)_o = 8125$ ft/sec; $h_o = 601\ 600$ ft.

Figure 14.- Continued.



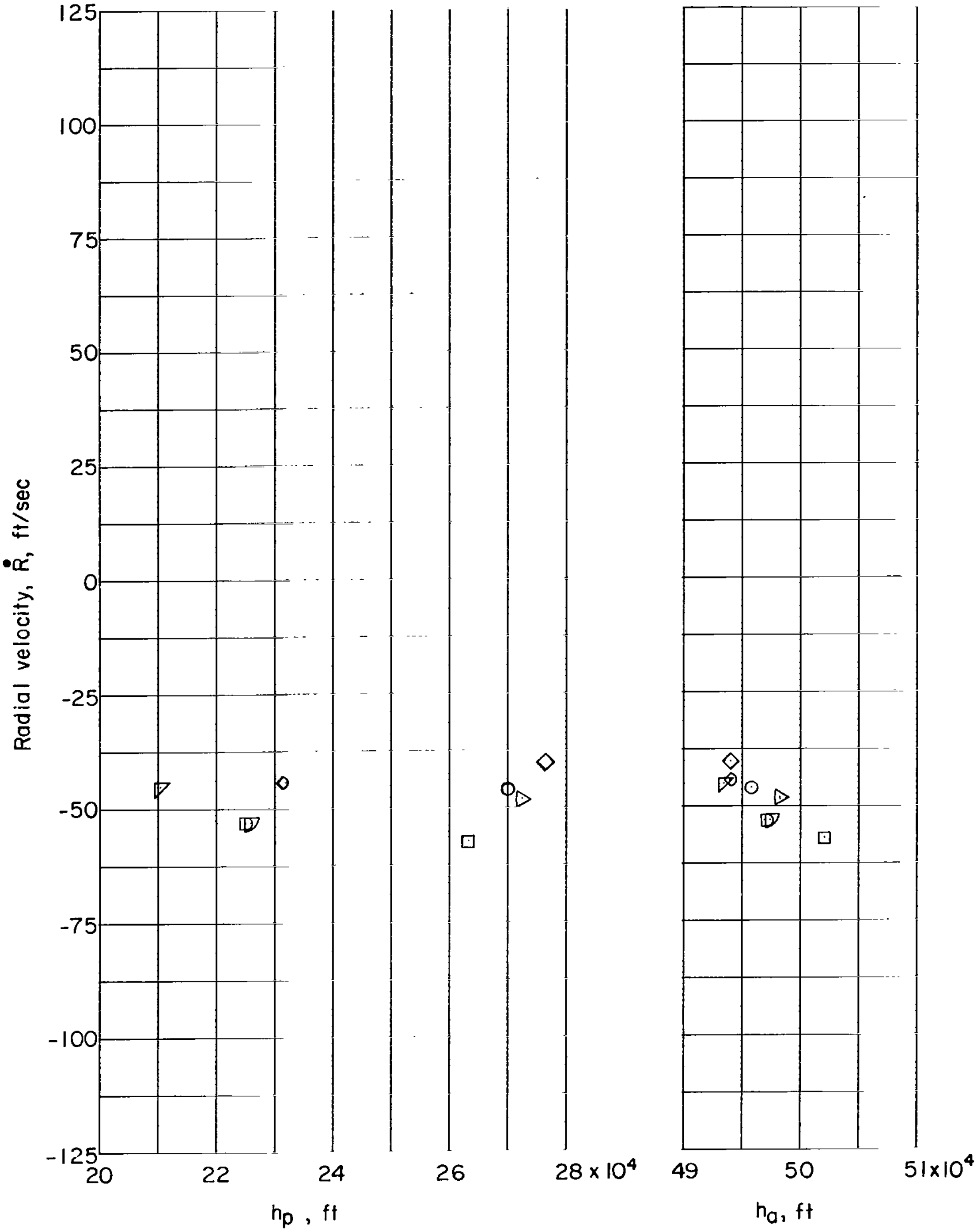
(b) Concluded.

Figure 14.- Continued.



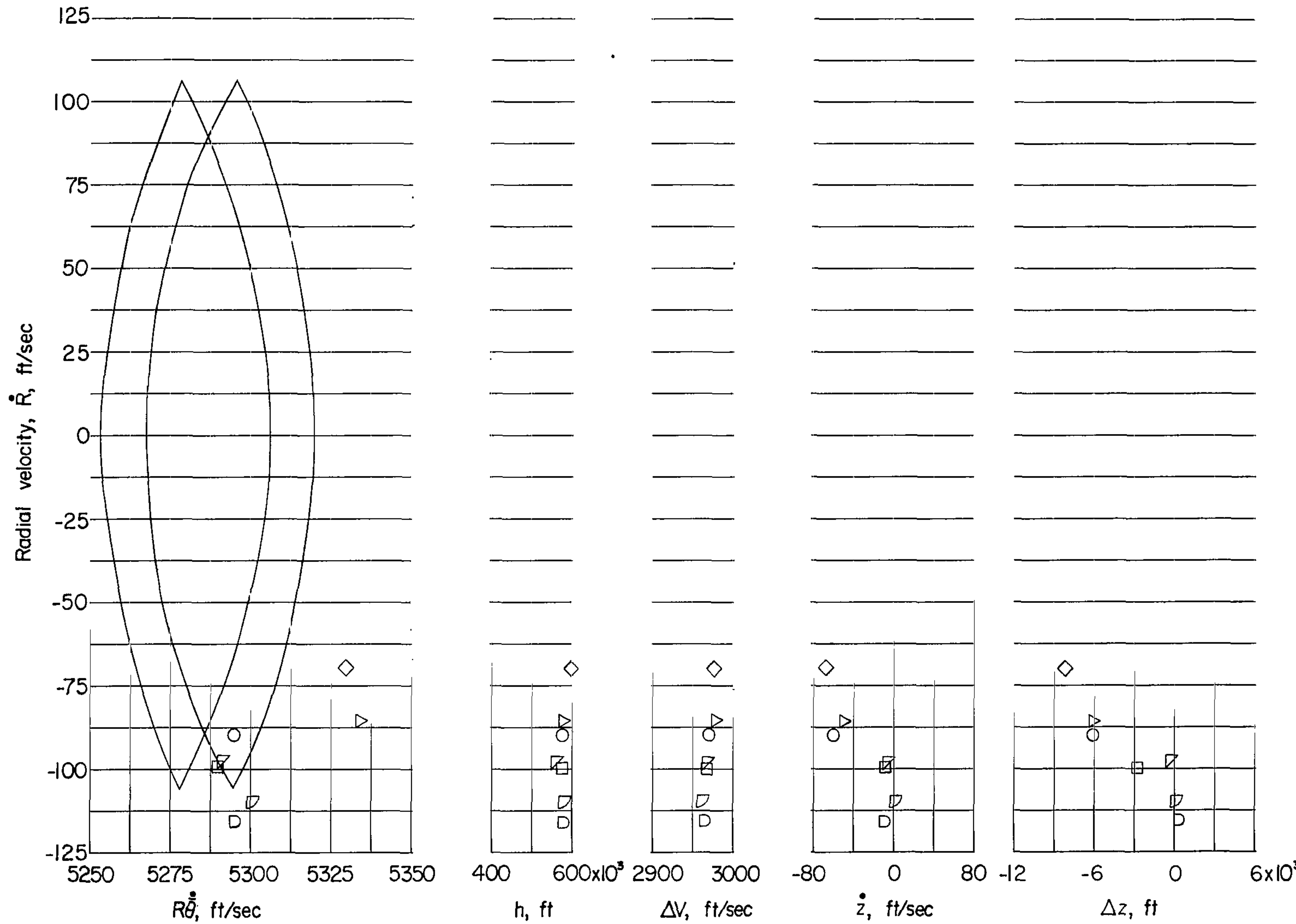
(c) $\dot{R}_o = -1138$ ft/sec; $\left(R\dot{\theta}\right)_o = 8075$ ft/sec; $h_o = 601\ 600$ ft.

Figure 14.- Continued.



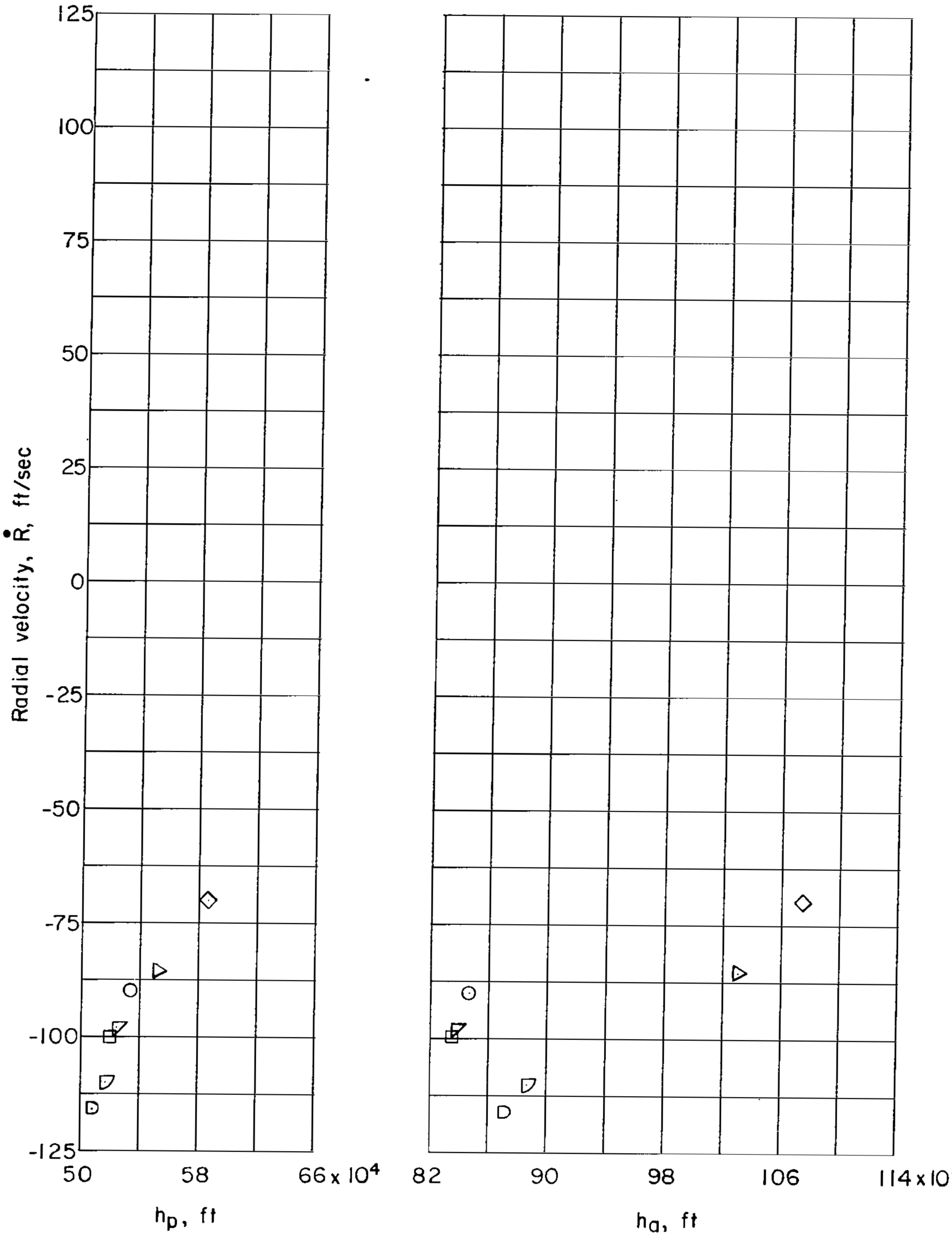
(c) Concluded.

Figure 14.- Continued.



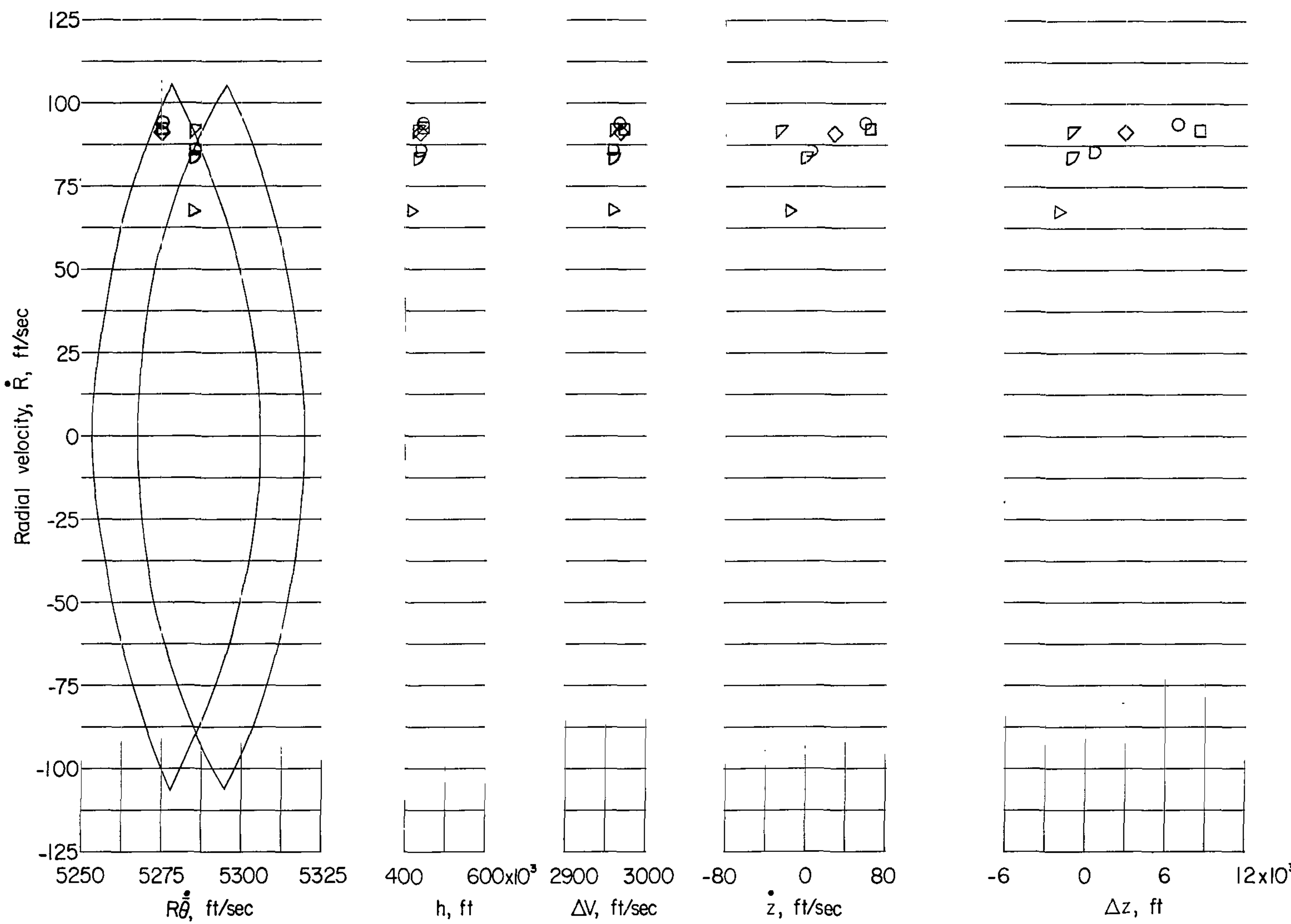
(d) $\dot{R}_o = -1138$ ft/sec; $\left(R\dot{\theta}\right)_o = 8125$ ft/sec; $h_o = 701\ 600$ ft.

Figure 14.- Continued.



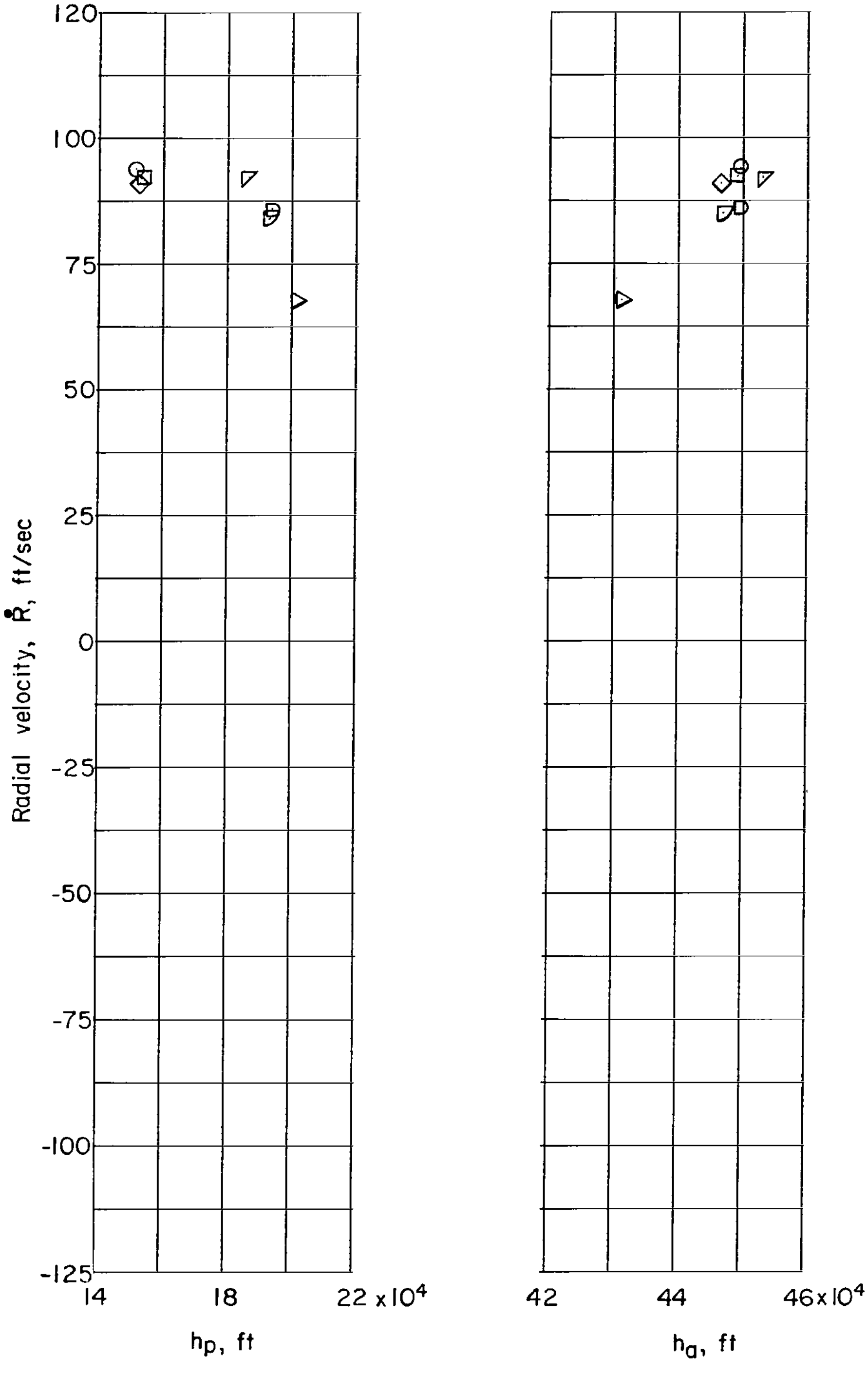
(d) Concluded.

Figure 14.- Continued.



(e) $\dot{R}_o = -1138$ ft/sec; $\left(R\dot{\theta}\right)_o = 8125$ ft/sec; $h_o = 501\ 600$ ft.

Figure 14.- Continued.



(e) Concluded.

Figure 14.- Concluded.

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—NATIONAL AERONAUTICS AND SPACE ACT OF 1958

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